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ASSESSMENT OF MANNED LUNAR
MISSION TECHNOLOGICAL S[illegible]
(Preliminary Report)

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by

[illegible] J. Olmer, and [illegible]

20 March 19[illegible]

Prepared for
National Aeronautics and Space Administration
under Contract NASw-340

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... Plans for the Launch Vehicle Technology Support Prog ... ... to
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jectives of the manned lunar missions, and (c) identification of cri ... l
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provide timely support for the progr... m obj... ... es of the manned lun... mi
sions.

T... technical areas discussed in this report are Astronautics, En
vir... nen... Effects, Guidance, and Control; the technical areas of Com-
m... ati... , Structural and Material Design and Fabrication, and Elec-
... Power Supplies and Distribution are to be covered in subsequent re-
... . The technical areas of Data Processing, Instrumentation and Test
... Applied Physics and Mathematics will not be analyzed in this contract...
...; it is recommended that the analysis of these latter technical areas be
...rtaken.

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SUMMARY

[illegible] represents [illegible] contribution by [illegible] purpose of [illegible] is to assess [illegible] program [illegible] requirements [illegible] this objective [illegible] [illegible] with [illegible] [illegible] planned [illegible] additional [illegible]

[illegible] results are presented [illegible] involved in the manned [illegible] program. Four of [illegible] technical areas

[illegible],

Environmental Effects,

Guidance,

and Control.

[illegible] covered in this report. The remaining areas will be covered [illegible] reports.

Each section includes: (a) The analysis of the technical [illegible] (b) An assessment of relevant NASA support program. (c) The problem[illegible] not covered by NASA activities or by state-of-the-art.

The results obtained [illegible] indicate that a substantial [illegible] will require [illegible] CRI proposes to submit [illegible] tive [illegible] compatible with [illegible]A space program upon the [illegible] as an obligation.

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ASTRONAU[illegible]
TECH[illegible]CAL AR[illegible]
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TECHNOLOG[illegible] SUPPORT [illegible]UATION

APP[illegible]DIX A: BIBLIOGRAPHY

APP[illegible]DIX B: SECONDARY MISSION DEFINITIONS

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[illegible] Engine-Cu[illegible]
[illegible] . . . .

[illegible] . . . .

[illegible] Trajectory to [illegible]
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Lunar Mission [illegible] . . . .

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I. INTRODUCTION

1.1 One of the most pressing objectives of the National Aeronautics and Space Administration is the successful accomplishment of manned exploration of the moon within the not too distant future. To ensure effective and well coordinated planning and direction of the launch vehicle program of this manned lunar mission, it is important that comprehensive definitions exist which will provide an understanding of the program objectives. Also, the supporting technology program must be a sound base upon which future requirements of the launch vehicle program can be met. The demands placed on the technical areas of the support program should be compatible with the state-of-the-arts. It is the purpose of the study, discussed herein, to provide the Advanced Development Section of the Office of Launch Vehicle Programs with (a) a set of Technical Area Plans spanning the period 1960-1975, which will include a set of program objectives and statements of specific problems which must be solved in research and development, (b) an assessment of the applicability to the manned lunar mission of presently funded and supported projects in the launch vehicle technology program, and (c) an identification of critical gaps in the technology program. The effort is being accomplished under NASA contract NASw 340, dated 18 October, 1961.

1.2 The technical areas under analysis by this contracted study are:

a. Astronautics.

b. Environmental Effects.

c. Guidance.

d. Control.

[redacted]

e. Communication.

f. Materials and Structure Design and Fabrication.

g. Electrical Power Sources and Distribution.

h. Data Processing.

i. Instrumentation and Test.

j. Applied Physics and Mathematics.

The technical area of Propulsion was not to be covered by this study.

1.3 This report is an interim (preliminary) report; it is being forwarded to NASA for review and informal approval. Only four technical areas are analyzed herein: Astronautics, Environmental Effects, Guidance, and Control; all technical areas will be discussed in the final report. The discussions, plans, and recommendations presented in this report are subject to change; the analyses of these technical areas are not complete and will continue until completion of the contract.

1.4 The immediate interest of NASA is in the first generation vehicles of the period 1960-1975, such as the Saturn C-1, Saturn C-5, and the NOVA vehicles, which are to be used in the initial manned lunar missions. The analyses of the technical areas and the technology support program are directed toward the requirements of these first generation vehicles to accomplish the various missions. This study considers only research and study (no developmental) effort associated with subsequent generation vehicles; vehicles capable of (1) accomplishing futuristic missions not considered in this study, or (2) undertaking missions considered within the scope of this study, but which utilize techniques well beyond current and projected state-of-the-arts of the time period under question.

1.5 Each technical area was thoroughly investigated; extensive literature searches were made for applicable information. The NASA Library, ASTIA, the Library of Congress, and the ORI Library were used as primary sources of information in assessing launch vehicle state-of-the-arts and requirements associated with the manned lunar missions. Appendix A is a bibliography of the reports utilized and referred to in the contracted study. Only published reports were utilized in the study; sources of pertinent unpublished information were not contacted because of the lack of time and the lack of authority to undertake such endeavors.

1.6 The initial ultimate mission of the manned lunar program is the soft lunar landing and safe return to earth of a manned vehicle. Apollo is the first attempt toward this initial goal. For preliminary lunar mission flights, the primary flight missions include cislunar, circumlunar, lunar orbit, and lunar land categories.

1.7 Figure 1 outlines the possible combinations and permutations of flight sequences for the manned lunar missions. Figure 2 outlines, in the judgment of ORI, the probable mission flight sequences for the first generation vehicles. The sequencing reflects direct and interrupted flights; the latter involves orbital rendezvous and docking of the affected vehicles. The sequences have been combined into groups forming tasks, or secondary missions; the technical area analyses were based on the requirements of the launch vehicles in undertaking these missions. These secondary missions are:

a. Earth Launch and Orbit.

b. Orbital Rendezvous.

c. Orbital Docking.

d. Orbital Transfer, Assemply, Repair, Maintenance and Checkout.

e. Earth Orbital Launch and Translunar Flight.

f. Lunar Orbit and Landing.

g. Lunar Launch and Transearth Flight.

h. Earth Reentry and Land.

1.8 It is possible to analyze the state-of-the-art on the basis of the requirements for each task or more specifically in terms of the technical areas involved in each task. This procedure permits the orderly discussion of the results with a minimum amount of repetition. These tasks or secondary missions are defined in Appendix B.

1.9 Other lunar missions exist beyond the initial manned lunar missions analyzed herein. Figure 3 indicates probable spatial missions which will follow a successful Apollo mission. These missions may be able to utilize all or parts of the first generation vehicles discussed herein; perhaps second or third generation vehicles will be required. An analysis of launch vehicle requirements associated with these subsequent missions should be conducted. The state-of-the-arts within the appropriate technical areas should be investigated to establish compatibility with requirements. Although Apollo is only in the initial phases of development, planning of the technological support program for subsequent missions should be undertaken now to ensure that these missions can be successfully accomplished without undue delay.

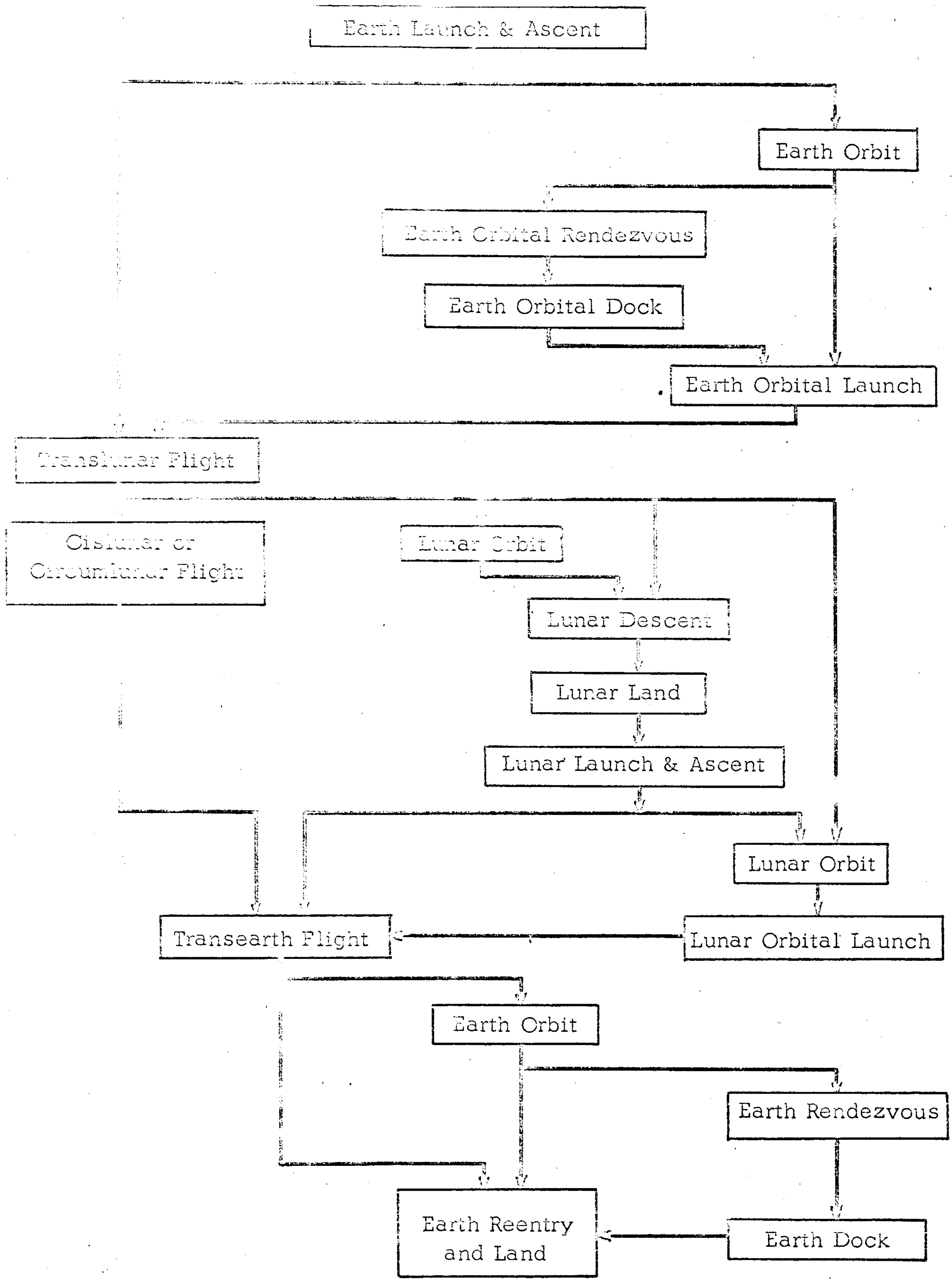


FIGURE 1. MANNED LUNAR MISSION FLIGHT SEQUENCES

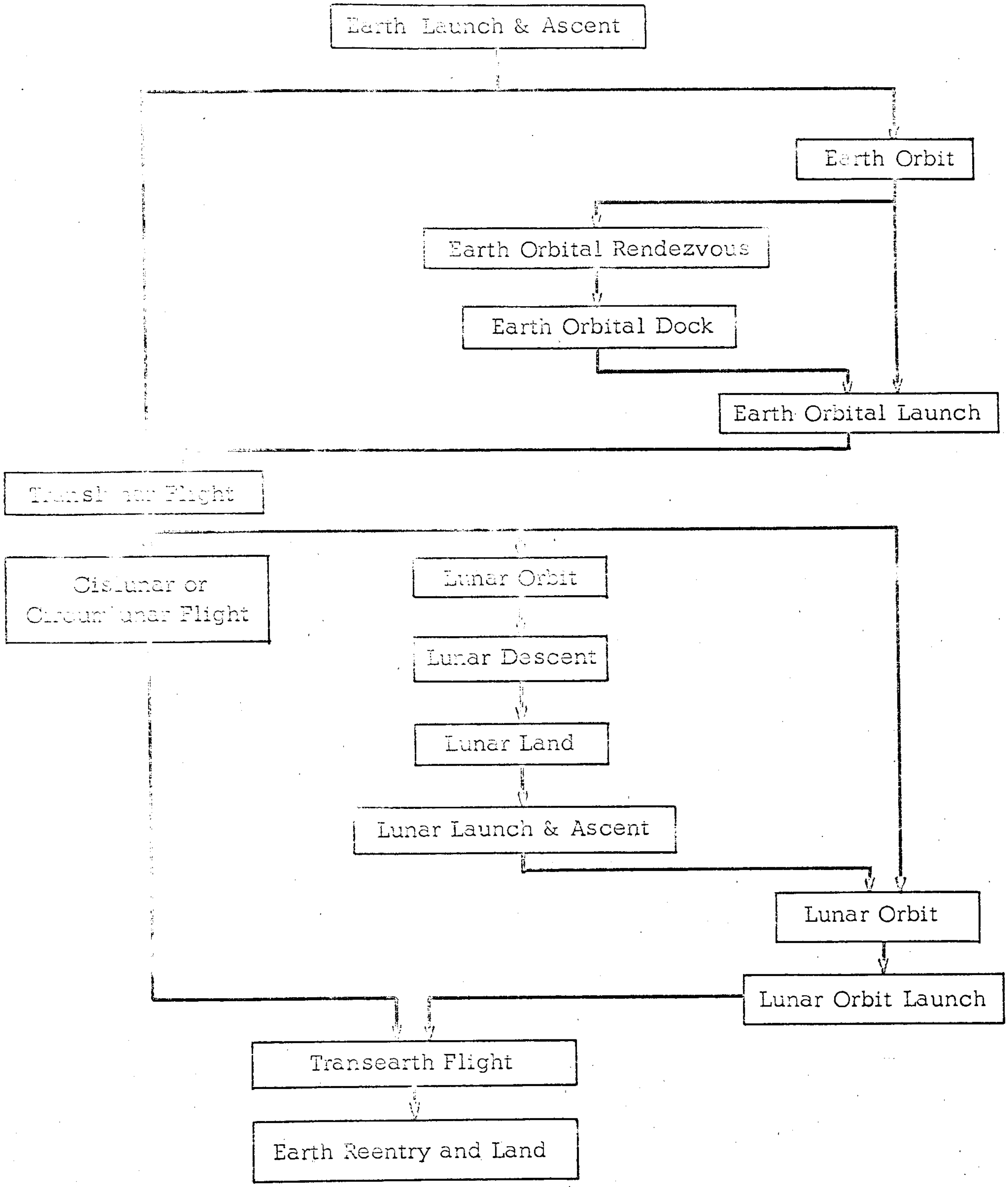
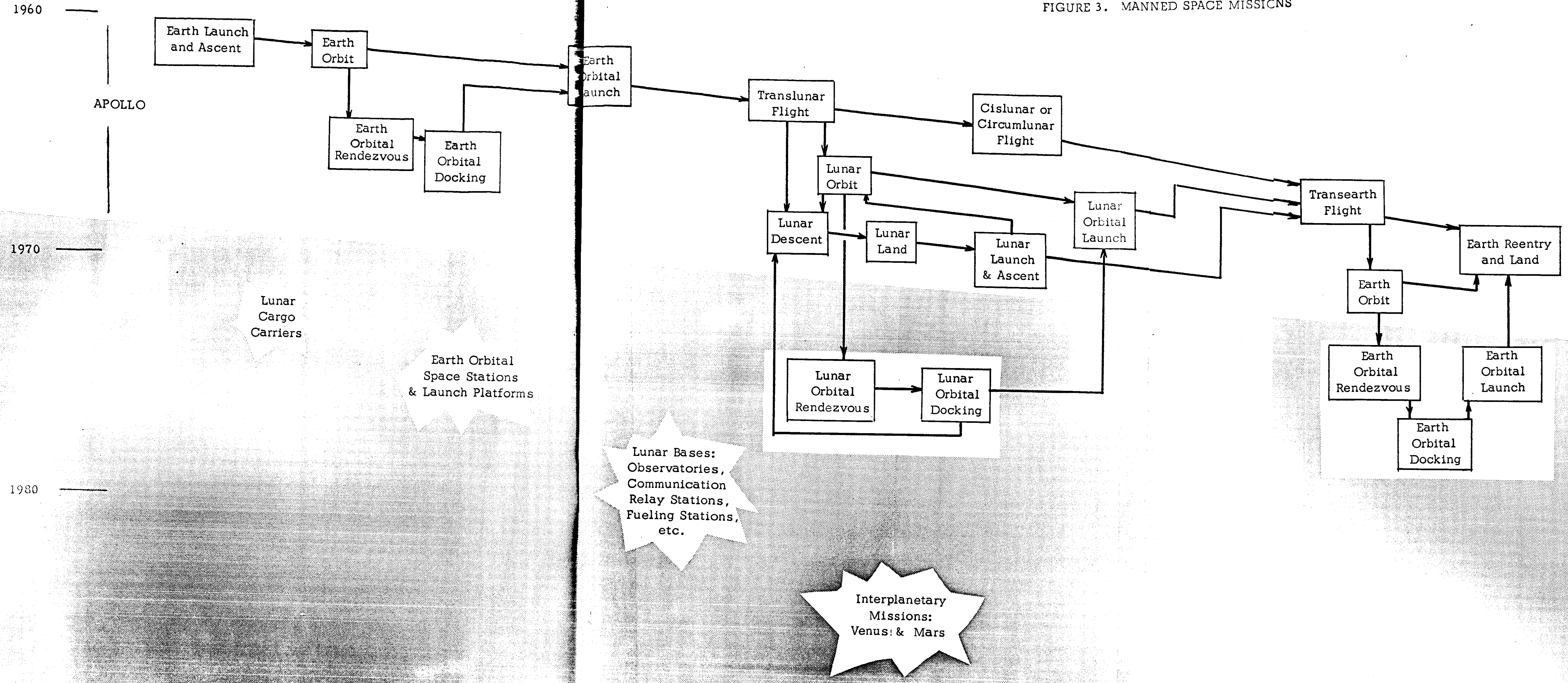


FIGURE 2. PROBABLE LUNAR MISSION FLIGHT SEQUENCES FOR FIRST GENERATION VEHICLES

FIGURE 3. MANNED SPACE MISSICNS



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1.10 Subsequent sections of the report discuss each of the technical areas; each section contains the respective Technical Area Analysis, the Technical Area Plan, and the Technology Support Program Evaluation. The analysis is a discussion of the technical area; state-of-the-arts and requirements associated with the various missions and technological incompatibilities are determined. The technical area plans which resolve the technological incompatibilities are presented as investigation or task sheets. These task sheets outline specific investigations to be conducted; each sheet includes a <u>Task Statement</u>, statements of <u>Justification</u>, <u>Present Status</u>, <u>Criticality</u>, and <u>Mission Applicability</u>, and presents a <u>Reference</u> to the problem as discussed in the Technical Area Analysis. In lieu of NASA Long Range Plans, the time element of the plan is indicated by mission applicability. The technology support program evaluations are presented in tabular form. The projects are separated into categories—a remark is made relative to each project—general remarks are presented relative to each category.

II. ASTRONAUTICS

2.1 This analysis considers the problems generated by the extra-terrestial nature of lunar missions. The areas involved include space mechanics and cosmology. Some of the problems to be considered concern the basic energy requirements for the various lunar missions; the accuracy to inject a vehicle into the proper trajectory for lunar circumnavigation or landing, as well as for safe return to the earth; celestial navigation, and so forth. Excluded are these problems involving functions performed exclusively by earth based installations.

2.2 It is evident that the problems under considerations can be evaluated only if the basic objectives of the lunar mission are defined. As a crude illustration, consider a mission aimed at scoring a first for national prestige. The need for quickly achieving the objective will obviously overshadow the basic accuracy, payload and costs desiderata, even perhaps, the reliability requirements for recovering the crew. The scientific returns may be slight, maybe a few blurred photographs as in the case of Lunik III. A bonafide scientific mission, on the other hand, will require more precise control of circumlunar trajectories or orbits as well as the recovery of the data. The problems in space navigation, communication, etc. involved in the two types of missions differ widely. The absence of criteria defining the mission, that is, the objectives of NASA lunar program, makes it difficult to define the technical requirements and compare them to present state-of-arts capabilities. It is clear that the task of defining the lunar mission criteria, perhaps in terms of trade off values between national prestige, costs and expected value of potential scientific or military returns cannot be undertaken within the scope of this program.

2.3 The following discussion is based on the official definition[1] of NASA as "an agency for the administration of a peaceful, non military space program . . . whose responsibility is to conduct the scientific exploration, investigation and utilization of space for peaceful purposes." On this basis, it becomes possible to define, at least in broad generalizations, the technical requirements for a "peaceful and scientific" lunar mission, and to indicate what problem areas remain to be investigated to fulfill this objective.

2.4 The most critical area of a manned lunar mission concerns the terminal phase of the trip: the reentry into the earth atmosphere and recovery of the crew. These terminal problems will be considered first.

Earth Capture of a Lunar Vehicle

2.5 A substantial portion of the kinetic energy of the inbound vehicle must be dissipated near the earth to prevent parabolic escape into a translunar or solar orbit.

2.6 Earth capture by retrothrust to below orbital velocity, say to 11,000 fps imposes excessive weight penalities at launch[2] and does not appear to be within the capability of present launch propulsion systems. Similar payload considerations exclude the use of high L/D ratio capsules, capable of extensive maneuvering within the atmospheric layer during which the excess kinetic energy is dissipated through aerodynamic braking.

2.7 Present lunar missions appear to be restricted to ballistic reentry vehicle with low (.5) L/D ratios. The limited capabilities of these vehicles severely restricts the angle of reentry within the atmosphere.

1/ D. D. Wyat, NASA, Assistant Director of Program Planning and Coordination, 20 January 1960.

2/ The only conceivable mission is a Nova and a Saturn C-5 launch of a 15,000 lb manned capsule with retrothrust capabilities for slowing to suborbital velocity. This mission is restricted to simple lunar flyback, the fuel required for lunar orbiting or landing exceeding the capabilities of the Nova. It thus appears that mission profiles including inbound rendezvous of the lunar vehicle with an earth orbiting platform cannot be considered seriously at present.

2.8 The limiting conditions, or reentry corridor of Apollo type capsules at parabolic velocity have been extensively studied.[3-6] For an L/D ratio of .5 and a maximum decceleration of 10 g's, the perigee altitude must be kept within about 15 to 50 miles, the exact boundaries depending on the weight/cross sectional area of the vehicle. Computer simulations of circumlunar flights indicate that these requirements can be met with present navigational, computing and control capabilities.

Recovery of the Vehicle

2.9 The maneuvering capabilities of the Apollo capsule in the atmosphere have not been determined. Design criteria for an L/D ratio of .5 permit only a limited selection of landing point by the pilot. The accessible landing area (foot print) appears to be an approximately elliptical area,[7-10] about 1600 miles long in the downrange direction and from 300 to 400 miles at its widest point.

2.10 Furthermore, the location of the footprint along the reentry great circle depends on the perigee altitude and, probably, on the reentry speed. The downrange location may vary from 500 miles to about 20,000 miles from the reentry point when the perigee varies

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3/ G. P. Edmonds, et. al., Trajectory Control for Reentry into the Earth Atmosphere, AF 33 (616) 3892 (MIT TR-198).

4/ D. R. Chapman, An Analysis of Corridor for Supercircular Reentry into Planetary Atmospheres, NASA TR R-55, R-11, 1959.

5/ P. Becker, et. al., Aerodynamics of Trajectory Control for Reentry at Escape Speeds, Astronautics Symposium, Paris, 1961.

6/ R. W. Ludens, Analysis of Atmospheric Entry Corridors, NASA TN D-590, 1961.

7/ L. D. Ely, Reentry Systems, Aerospace Corporation, 1961.

8/ A. Lees, et. al., "Use of Aerodynamic Lift During Reentry into the Earth Atmosphere," A.R.S. Journal, 29, p. 533, 1959.

9/ F. C. Grant, Dynamic Analysis of Simple Reentry Maneuvers for a Lifting Satellite, NASA TN D-47, 1959.

10/ E. C. Foudriat, et. al., Guidance and Control During Direct Descent in Parabolic Reentry, NASA TN D-979, 1961.

within the altitude range insuring safe reentry.11/ In other words, no single point of the reentry great circle can be reached from all portions of the reentry corridor. Recovery facilities must be dispersed half way around the globe even in the most favorable case where the plan of trajectory and the time of arrival are precisely controlled. Restrictions in the downrange spread of possible landing points requires improved control of reentry parameters; for example, landing within a specified 2000 mile arc requires that the perigee altitude be controlled within 1.5 miles. The downrange location of the footprint also depends on the value of the L/D ratio of the vehicle.12/

2.11 Finally, because of the rotation of the earth, the reentry plane of the lunar vehicle appears to move on the surface of the earth. Selection of a reentry great circle requires that the time at which the vehicle enters the atmosphere, that is, the duration of the moon-earth trip, be accurately controlled.

2.12 To summarize, any restriction in the permissible landing area increases the accuracy with which the reentry parameters (time of arrival, perigee altitude, magnitude and direction of reentry velocity vector) must be controlled. The relationship between the restrictions in reentry parameters and the desired landing area are extremely complex and depend, amongst others, on the latitude, shape and size of the area.13-16/ The relationships are further complicated

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11/ Preliminary Parametric Analysis for an Orbital Rendezvous Base System, Northrop Corp., Report ASG TM-61-59, 1961. (Date in this report refers to a vehicle of L/D = .65; reentry requirements may be more severe for a vehicle of lesser maneuvering capabilities.

12/ L. L. Levy, et. al., Comparison of Two Maneuvers for Longitudinal Range Control During Atmospheric Reentry, NASA TN D-1204, 1962.

13/ J. H. Lowry, "Control and Guidance of Point Return Vehicles," Proceedings on Guidance of Aerospace Vehicles, Boston, 1960.

14/ J. A. White, et. al., "Guidance of Space Vehicle to a Desired Point on the Earth Surface," A.R.S. Journal, January 1961; Preprint 61-41, Am. Astron. Soc., 1961.

15/ A. G. Boissevain, Effect of Lateral and Longitudinal Range Control on Allowable Reentry Conditions for a Point Return from Space, NASA TN D-1057.

16/ W. J. Pragluski, et. al., Lunar Trajectory Analysis, NASA Industry Apollo Technical Conference, Washington, 1961.

by the fact that the reentry parameters are not independent; for instance, the time of reentry cannot generally be changed without altering the trajectory, that is, the perigee and/or the reentry velocity vector.

2.13 Present NASA concepts for lunar missions[17] call for a permissible landing area extending almost halfway across the globe, from the western Pacific to the eastern Atlantic and passing over the continental U.S. The greatest width of this area, over the U.S. is about 1800 miles, corresponding to a permissible variation of $\pm 13^\circ$ in the orbital plane. It has not been possible to ascertain whether these requirements have been arbitrarily fixed or whether they represent the results of a planned analysis showing the optimum tradeoff between (a) our present capabilities in space navigation and (b) our capabilities for deploying sufficient land and sea facilities to insure fast and reliable recovery of the capsule.[18]

2.14 Much effort is being devoted by NASA[19-29] and others[30-37] to the solution of the problems involved in the reentry and point landing of

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17/ Project Apollo, Statement of Work, Phase A, NASA, 1961.

18/ Proceedings on the Recovery of Space Vehicles Symposium, Los Angeles, 1960.

19/ D. P. Harry, Analysis of Errors and Requirements for an Optimum Guided Technique for Approaches to Reentry with Interplanetary Vehicles, NASA TR R 102, 1961.

20/ NASA TN D-590, op. cit.

21/ D. R. Chapman, An Approximate Analytical Method for Studying Reentry into Planetary Atmospheres, NASA TR R-11, 1959.

22/ R. E. Sly, An Analytical Method for Studying the Lateral Motion of Reentry Vehicles, NASA TN D-325, 1960.

23/ J. W. Young, A Method for Longitudinal and Lateral Range Control for a Vehicle Entering the Atmosphere of the Rotating Earth, NASA TN D954, 1961.

24/ D. C. Cheatham, et. al., The Variation and Control of Range Travelled by High Drag Variable Lift Entry Vehicle, NASA TN D-230, 1960.

25/ J. M. Eggleston, et. al., Trajectory Control for Vehicles Entering the Earth Atmosphere at Small Flight Path Angles, NASA Memo 1-19-59L; TR R 89, 1959.

26/ F. C. Grant, Modulated Reentry, NASA TN D-452, 1960.

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27/ A. Assadourian, Longitudinal Range Control During the Atmospheric Phase of a Manned Satellite Reentry, NASA TN D-253, 1960.

28/ R. C. Sommers, Point Return from a Lunar Mission, NASA TN D 1192, 1961.

29/ Proposed NASA Projects: Reentry and Return Guidance Studies.

30/ L. D. Ely, op. cit.

31/ L. R. Bush, Study of Accurate Reentry and Precision Landing of Orbital Earth Satellites, Cornell Aeron. Lab. VF 1351 H-1.

32/ L. M. Gaines, Optimum Approach and Landing Techniques for Manned Reentry, Inst. Aerospace Sciences Inc. 115, p. 1809, 1961.

33/ A. Ferri, et. al., Practical Aspects of Reentry Problems, PI BAL Report 705, 1961.

34/ K. Wang, Reentry Trajectories with Aerodynamic Faces, PI BAL Report 647, 1961.

35/ R. H. Smith, "Supercircular Entry and Recovery with Maneuverable Vehicles," Aerospace Eng. 20, p. 12, 1961; IAS Paper 61, 114, 1808.

36/ L. W. Warzecka, (GE), "Performance and Design Considerations for Maneuvering Space Vehicle Return to Earth Surface," Inst. of Aeronautical Sciences, Paper 50-59, 1960.

37/ Lockheed, Northrop, General Dynamics, etc.

the inbound capsule. The scope of most of these investigations, however, is limited to orbiting satellites; no attempt has so far been made to integrate the results into the over-all lunar mission concept. This point is illustrated by the following considerations.

2.15 The probability with which the capsule can be successfully recovered should be compatible with the reliability level assigned the other phases of the mission. The recovery reliability depends on such factors as the extent and disposition of air-sea-land recovery facilities, the speed with which these facilities can be transferred to the impact point, the possibility of predicting this impact point by radar tracking of the capsule during its terminal flight and, of course, the accuracy with which the potential landing area can be reduced by controlling the reentry parameters. It should be possible to define (through usual war gaming or Monte Carlo techniques) the conditions optimizing the reliability of the reentry-rescue operations. Investigation of the optimum disposition of rescue facilities[38-40] have been made only in reference to the recovery of orbiting satellites. Our experience with the first manned orbital Mercury should indicate the difficulties to be expected in recovering a lunar vehicle whose reentry parameters cannot be closely controlled. Although this phase of the problem lies outside the scope of this investigation, it is clear that the recovery problem should be treated as a whole, that is, the nature and disposition of surface facilities should determine the guidance requirements in space and vice versa.

2.16 Lack of information along these lines leads one to suspect that the permissible landing area requirements spelled out in NASA statement of work do not represent the results of such optimizing technique but reflect, rather, the arbitrary estimate of the area which can be adequately surveyed.

2.17 It was pointed out earlier that any limitation in the landing area imposes constraints on the reentry parameters. The latter, in turn, affect the requirements for space navigation and midcourse corrective maneuvers. It is thus essential to translat NASA landing requirements in terms of boundaries in reentry parameters in order to properly examine the problems in astronautics and their implication on the over-all lunar mission. An attempt to do so is presented in the next section.

38/ Proceedings of the Recovery of Space Vehicle Symposium, Los Angeles, 1960.

39/ J. S. Hamilton, Satellite Water Recovery Feasibility, ABMA, DLMT 4, 59, 1959.

40/ M. A. Fischl, (GE), Problems of Visual Search in the Recovery of Space Vehicles, American Rocket Soc., Inc., 808-59, 1959.

[redacted]

Astronautics

2.18 With chemical propulsion, most of the energy required to propel a vehicle into space is generated within a short period at launch.[41] The remainder of the flight is essentially unpowered, the vehicle coasting in the complex gravity field of space. The trajectory is mainly controlled by the magnitude and direction of the thrust vector at launch. Slight errors in controlling this vector cause the space-time coordinates of the vehicle to diverge from the desired course, all the more so as the length of the trip increases.

2.19 A discussion of launch accuracy requirements cannot be conducted within the scope of this study. This would require analysis of trajectories of various energy and consideration of a wide range of azimuth and injection flight angles. To illustrate the following discussion, a few typical values of launch accuracy requirements was extracted from the literature[42-50] and are summarized in Table 1.

41/ Nuclear and ionic propulsions may, eventually, provide means for sustained power space flight.

42/ H. A. Lieske, "Accuracy Required for Trajectories in the Earth Moon System," Vistas in Astronautics, Vol. I, 1959, also RAND P-1022, 1957.

43/ Far Side Preliminary Studies, Aeronutronic Systems Inc., TN 58-653, 1958.

44/ R. H. Grube, "Terminal Guidance for Lunar Probes," Proceedings on the Guidance of Aerospace Vehicles, Boston, 1960.

45/ "Astronautics and its Applications," Staff Report of the Selected Committee on Astronautics and Space Exploration, House of Representatives, 85th Congress, 1959.

46/ Study of Large Launch Vehicle Subsystems, North American Aviation, SID 61-327, 1961.

47/ Preliminary Analysis for an Orbital Rendezvous System, Northrop Co., ASG-TM-61-59, 1961.

48/ Trajectory Studies, RAND RM-1728, 1956.

49/ G. Shapiro, "Trajectory Problems in Cislunar Space," Proceedings on Guidance of Aerospace Vehicles, Boston, 1960.

50/ H. J. Gordon, A Study of Injection Guidance Accuracy as Applied to Lunar Missions, Jet. Prop. Labs, TR 32-90, 1960.

[redacted]

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TABLE 1

LAUNCH ACCURACY REQUIREMENTS FOR LUNAR MISSIONS

	ΔV (fps)	$\Delta\gamma$ (degrees)	Δt (seconds, from a 90 minute orbit)
1. Passage in vicinity of moon			
Perilune within ± 15,000 mi	150	2.5	42
Perilune within ± 100 mi	4	.02	.3
Perilune within ± 30 mi	1	.005	.07
2. Hit anywhere on moon			
Within 100 miles from aimpoint	75	.5	7
Within 30 miles from aimpoint	2	.01	.15
Within 1 mile from aimpoint	.5	.0001	.015
3. Perilunar transit with return			
Within a 15 mi reentry corridor	.7	.001	—
Within a 3 mi reentry corridor	.1	.0002	—
4. Lunar launch with return			
Within a 15 mi reentry corridor	.8	.25	.5 (for 116 minutes lunar orbit)
State-of-the-Art:			
5. ICBM launch accuracy, with radioguidance environment and control verniers (1σ)	5	.05	—
6. Centaur launch accuracy (1σ)	4	.02	—

[redacted]

[redacted]

2.20 The results[51-53] of Table 1 are expressed in terms of the magnitude Δv and direction $\Delta \gamma$ (within the lunar orbital plane) of allowable errors in the velocity vector at burn-out. Errors in direction ($\Delta \gamma$) have been translated in terms of launch window (Δt) for earth or lunar orbital launches. The Δv and $\Delta \gamma$ tolerances are mutually dependent; the greater the allowable velocity error, the less the directional error and vice versa.

2.21 State-of-the-art in launch accuracy is summarized in the last two lines of Table 1. It is clear that present capabilities are incapable of insuring the safe return of a manned lunar vehicle, that is, of preventing either its catastrophic decceleration in the earth atmosphere or its permanent escape into space. Furthermore, the results of Table 1 must be considered as optimistic estimates, because:

a. They do not take into consideration angular launch errors outside the lunar orbital plane. Three dimensional trajectories have been the subject of very few investigations,[54-57] probably because of computational difficulties. It is recalled that three dimensional (dog leg) trajectories may have to be used to control the reentry orbital plane.

51/ The data in the literature vary by a factor of 2 or 3. The discrepancies are traced to differences in the underlying assumptions and in the approximations made in computing the 3 body trajectories.

52/ The accuracy requirements vary in a complex fashion with the burn out speed, the nature of the trajectory (direct, retrograde), the launch conditions and so on. The results in the table must, therefore, be considered as approximations to be used for comparing the mission profiles.

53/ The very severe requirements of unguided perilunar missions with return to earth reflect the sensitivity of the orbit to the perilune altitude. The allowable errors for lunar launch are much less severe.

54/ W. H. Michael, et. al., Three Dimensional Lunar Mission Studies, NASA 6-29, 1959.

55/ A. B. Mickelwait, Analysis and Numerical Studies of Three Dimensional Trajectories to the Moon, Space Techn. Labs., GM TM-0165-00287, 1958.

56/ W. H. Michael, Effect of Eccentricity of Lunar Orbit, Oblateness of Earth, Solar Gravitational Field on Lunar Trajectories, NASA TN D-227, 1960.

57/ J. N. Nielson, "Three Dimensional Satellite Orbits with Emphasis on Reentry, Dynamics and Oblateness Effects," Aerospace Eng., Vol. 18, p.60, 1959, NASA 12-4, 1958.

[redacted]

b. Few studies[21] take into consideration minor perturbations due to the oblateness of earth and moon the variations of the earth-moon barycenter, the solar gravitation and so on.

c. Unavailability of surface tracking, computing and checkout facilities on the moon may be expected to degrade the accuracy of lunar surface or orbital launches, at least within the initial phases of the lunar program. Realistic evaluation of lunar launch accuracy is not presently available.

d. The state-of-the-art indicated in Table 1 is valid only when launch occurs within the line of sight of earth tracking stations, that is, when radio communication can be maintained with the vehicle. Launch from earth parking orbits are expected to have lower performances; lunar launch still less accuracy.

2.22 To summarize, manned lunar missions will require midcourse corrective maneuvers to compensate for the almost inevitable errors at launch. Such corrections, however, impose launch payload penalties resulting in additional weight of the fuel, navigational, guidance and communication equipments which must be carried in the vehicle. It is essential to minimize these penalties because of the limited capabilities of present boosters. The interactions between the elements of the recovery problem and midcourse corrections are considered in the next section.

Midcourse Corrections

2.23 The preceding section indicated the need for midcourse corrections in a manned lunar mission. The following considers some of the problems associated with the determination and application of these corrections. The problems include:

a. The determination of the space time coordinates of the vehicle (navigation).

b. The computation of deviations from the nominal trajectory and velocity corrections to be initiated (computation).

[21] J. P. deVries, Generalized Interplanetary Trajectory Study, GE Report R 60 SD 465, 1961.

c. The evaluation of the magnitude of midcourse corrections and resulting penalties at take off (launch penalties).

d. The optimum sequencing of these corrections.

e. The accuracy with which the corrections can be applied in space (control).

2.24 It should be clear that the errors in each of these areas must be compatible; for instance, there is little point in being able to determine accurately the position of the vehicle and the required course corrections if the thrust cannot be controlled with sufficient accuracy to actually decrease some of the trajectory errors.

2.25 Determination of the Vehicle Space Coordinates. One of the Apollo design requirements is for on-board command of the vehicle, which implies the ability to perform all phases of the mission without the use of intelligence transmitted to the vehicle.59/

2.26 It has been suggested that earth-based stations may track the vehicle and transmit command for proper corrections. This so called command guidance will not stand the scrutiny of even lunar navigation requirements. The relative position of trajectory with respect to observations points during the relation of the earth and the required accuracies and the unreliability of space communications make any optical or radar tracking from the earth impractical. A number of techniques and systems have been proposed 60-62/ to determine the space-velocity coordinates of the vehicle. These involve, basically the optical measurement of the angles subtended between the earth, the moon and one or more fixed stellar directions. The accuracy with which these angles can be measured is

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59/ Y. C. Lees, "Some Considerations Pertaining to Space Navigation," Vistas in Astronautics, Vol. 2, 1959.

60/ Analysis of Guidance Control and Stabilization Systems, Martin Co., Report 8963, 1957.

61/ J. W. Unger, On the Midcourse Navigation for Manned Interplanetary Space Flights, DSP TR 258, 1958.

62/ L. Larmore, "Celestial Observation for Space Navigation," Institute of Aeronautical Sciences Meeting, Los Angeles, 1958.

estimated[63-67] at a fraction of a minute of arc, corresponding to position error in space of some 40 miles, (σ), the exact value depending on the technique used, the stars selected and the location of the vehicle relative to the earth or moon. The basic technique is to be supplemented by optical or infra-red measurement of the apparent diameter of the earth or moon and by radar altimetry in the immediate vicinity of these celestial bodies so that the error in position is expected to decrease to a few miles as the vehicle approaches either the earth or the moon. In the more sophisticated systems, the space coordinates are repeatedly measured and the errors smoothed out by automatic computation.

2.27 No reliable technique for directly measuring the instantaneous velocity of a space vehicle has been reported in the literature.[68] Velocity is to be derived from the rate of change of the space coordinates (vide infra). It is curious to notice that few systematic evaluation[69] of errors in computing the velocity is reported in the literature. It is recalled that even small velocity errors have large effects on the future position of the vehicle relative to that of moving target (moon or earth) and, consequently affect the location of the lunar or earth landing point.

Computation of Corrections

2.28 It is generally accepted that midcourse corrections will have to be computed onboard. While it is theoretically possible to transmit successive position data, as measured by the vehicle, to an earth computing center, and to receive correction orders, it is estimated that the reliability

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63/ A. F. Bowen, Guidance and Control for Lunar Missions.

64/ S. F. Schmidt, et. al., A Study of a System for Midcourse Navigation.

65/ NASA-Industry Apollo Technical Conference, 1961.

66/ J. A. White, A Study of the Effects of Errors in Measurement of Velocity and Flight Path on the Guidance of Space Vehicles Approaching the Earth, NASA TN D 957, 1961.

67/ NASA TR R 102, op. cit.

68/ Measurement of the Doppler shift of Lyman α radiation has been shown to have insufficient accuracy; W. J. Haywood, "Application of Optical Techniques to Interplanetary Navigation," Inst. Aerospace Sciences, Inc., 115, 1827, 1961.

69/ NASA TN D 957, op. cit.

[redacted]

of radio communications in space is not at present compatible with the over-all reliability assigned to a manned lunar mission. Interruption of communications may result not only from equipment failure but from unfavorable locations of the space vehicle (behind the moon) or earth station (behind the earth), as well as from unexpected space phenomena (solar flares).

2.29 A presently accepted concept for midcourse corrective maneuvers consists in feeding a computer with the successive measurements of the vehicle position; smoothing the variations and determining the variance between results; assigning a weight to each set of data as a function of its departure from the mean; determining the actual trajectory and/or the predicted miss distance; selecting from a stored memory the minimum energy orbit which will reduce the predicted miss to zero and, finally, determining the corrective thrust to place the vehicle on this new orbit.

2.30 The problems involved in the determination of trajectories in the earth moon systems[70-73] and in the computation of the minimum energy, mid-course corrective maneuvers[74-77] are well covered in the literature only part of which can be reproduced here.

70/ R. J. Weber, et. al., Lunar Trajectories, NASA TN D-866, 1961.

71/ O. R. Burg, On the Solution of Statistical Problems in Trajectory Analysis, LMSD, 704-206, 1960.

72/ R. H. Tolson, Effect of Some Typical Geometrical Constraints on Lunar Trajectories, NASA TN D 938, 1961.

73/ H. A. Lieske, Circumlunar Trajectories Study, RAND P 1441, 1958. Statistical Trajectory Studies, Proposed NASA Program.

74/ A. L. Friedlander, Study of Statistical Data Adjustment and Logic Techniques as Applied to Interplanetary Midcourse Guidance Problems, NASA TR R-113, 1961.

75/ J. Lorell, "Velocity Increments to Reduce Target Miss on Coasting Trajectories," Advances in Astronautical Sciences, Vol. 6, 1961.

76/ P. T. Smith, Discussion of Space Vehicle Guidance Problems, RAND, P1568, 1958.

77/ T. J. Wong, The Effect of... Guidance Requirements for the Return Lunar Flight, NASA TR R-80, 1960.

[redacted]

2.31 It is estimated[78-80/] that the operations involved in computing the midcourse corrections can be fulfilled by a digital computer with a storage capacity of 10,000 (24 bits) words. This should be adequate to cover the segmental display for 3 bodies simulation, the star catalog, the attitude control and the like. Such equipment may not weigh over 80 lbs and appear to be well within our present capabilities.

2.32 However, a number of problems in the computation of midcourse corrections have received insufficient attention. These problems briefly discussed below, include:

a. The extension of midcourse corrections in the three dimensional space.

b. The control of time of reentry into the earth atmosphere.

2.33 Around year 1970, the axis of rotation of the earth will make an angle of $90 - 18 = 72^{\circ}$ with the lunar orbital plane. The trajectory of a vehicle injected within this plane is, of course, two dimensional. The possible landing area on the earth is limited to a tropical zone between latitudes 18°N and 18°S.

2.34 If the landing area is to include the continental U.S., as specified in NASA statement of work, the incoming vehicle must depart from the lunar orbital plane at some point during the moon-earth leg of the trip. From this point on, the trajectory becomes three dimensional.

2.35 The midcourse correction problems considered in the literature have so far been limited to two-dimensional trajectories. It should be desirable to extend the investigation to the third dimension, if only to prove that constraints imposed by the determination of the landing area do not introduce problems exceeding the capability of the proposed navigation and compuation techniques.

78/ A. F. Bowen, A Guidance and Control Concept for Lunar Missions, NASA Industry, Apollo Technical Conference, Washington, 1961.

79/ Development of High Speed Digital Computers for Space Navigation Program, DA-04-495 ORD-1696, 1960.

80/ D. H. Blauvelt, et. al. "The Role of Computers in Aerospace Vehicles," Proceedings, IAS Meeting, Orlando, 1961.

2.36 The possible landing points of an earth bound vehicle are located on the intersection of the earth surface with the plane of trajectory.[81] Because of the earth rotation, the location of the reentry great circle depends on the time of reentry. Near the equator, each hour delay in arrival causes a 1000 mile westward shift in the location of the landing locus. If the vehicle is to land within a predetermined area, the time of arrival must be controlled.

2.37 The literature quoted earlier refers only to the control of the spatial coordinates of the reentry point (or perigee). It has not been possible to find an analysis of the problem involving control of both the location and the time or reentry. Undoubtedly, control of the reentry time imposes additional constraints to the accuracy with which the take-off must be controlled and/or to the extent of midcourse corrections. The problem is compounded in the case of a circumlunar mission, in which small trajectory errors such as the duration of the trip or the time of arrival grosely affect the pericynthion.[82]

2.38 It is essential to analyze the impact of earth and lunar launch errors on the time of reentry on earth, with the purpose of determining the midcourse corrections required to satisfy predetermined landing restrictions. The results may well lead to a redefinition of the concept of the recovery phase of the lunar mission; for example, it has been advanced[83] that present space guidance capabilities allow for a close control (within .5 miles) of the perigee altitude. If this is confirmed, it should be possible to land an Apollo capsule at a predetermined point on the reentry great circle taking into consideration the low range maneuvering capabilities. Variations in reentry time would only displace the longitude of this point so that the possible landing area would be a zone of constant latitude. The disposition of recovery facilities would, in this case, differ essentially from that indicated in NASA statement of work.

2.39 The third area deserving investigation concerns the determination of the magnitude of midcourse corrections.

81/ Neglecting for the moment the lateral maneuverability of the vehicle within the atmosphere.

82/ Nearest point from the moon.

83/ A. F. Bowen (loc. cit.)

2.40 The velocity increments required for safe reentry into the earth atmosphere range from 39 fps[84] to more than 1500 fps,[85] the majority of the estimates being in the vicinity of 200 fps.[86,87] While some of the discrepancies are traced to differences in the basic assumptions and navigational schemes, it is clear that definite estimates of fuel requirements must be obtained.

2.41 Furthermore, these results must be interpreted in the light of the following remarks:

a. None of the results considers corrections of trajectories in the three dimensional space nor corrections to control the time of arrival. These additional constraints may substantially increase the magnitude of corrections.

b. In some reports, the corrective velocity increments are computed on the basis of "average" errors at launch. In the manned mission, the recovery of the crew becomes a prime consideration and "average" values become meaningless. The frequency distribution of launch errors must be determined, in order to estimate the most unfavorable event associated with a probability of occurance compatible with the desired probability of succes of the mission. For example, if the launch errors are normally distributed, the 3σ error might well be taken as the basis for computing the magnitude of the required corrective thrust. On this basis, some of the data reported earlier may have to be revised upwards.

c. The data reported in the literature refers mostly to space missions or to lunar circumlunar flyback. Surface or orbital lunar launches have been considered in only one reference.[88]

84/ A. F. Bowen, op. cit.

85/ A. L. Friedlander, et al., Exploratory Statistical Analysis of a Planet Approach... NASA TN D471, 1960.

86/ S. G. Schmidt, et al., (Ames), A Study of a System for Midcourse Navigation, NASA Industry Apollo Technical Conference, 1961.

87/ Final Report, NASA Study of Large Launch Vehicles Subsystems, (North American Aviation) Report NASw-329, 1961.

88/ J. A. White, Study of the Effects of Errors...on the Guidance of a Space Vehicle Approaching the Earth, NASA TN D957, 1961.

[redacted]

It is unrealistic to expect that the accuracy of a lunar launch will duplicate that of an earth launch because of the unfavorable environment and the lack of surface tracking stations. The magnitude of corrective thrusts required in the earthbound leg of a lunar landing mission may thus be substantially larger than values indicated.

2.42 Furthermore, in lunar landing missions, each extra pound of fuel carried in the return leg of the trip requires additional energy expenditures during lunar take off; still more during lunar landing, and considerably more at earth launch. This pyramiding of take off weight is illustrated in Table 2 giving the earth take off penalties imposed by each 100 fps. corrective maneuver during the return leg of the trip.[89] for the various mission profiles described in the first column. The second and third columns list the nominal characteristic velocity for each mission, in the absence of corrective maneuver, and the theoretical weight of the vehicle at take off. The fourth column shows the extra weight at take off for each extra pound carried on the return trip. The fifth column indicates [90] the penalties corresponding to a total velocity correction of 100 fps., applied during the return leg to a 20000 lb vehicle. (Apollo capsule plus empty last stage).

2.43 Missions labelled "retrothrust reentry" refer to the slowing down of the inbound vehicle to suborbital speed (Δ v = 12000 fps) for better control of the reentry and landing problems. This type of mission, of course, should require only (at least theoretically) nominal midcourse corrections.

2.44 The results indicate that launch weight penalties imposed by midcourse corrections of a few hundred fps are only a few percents of the total

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89/ The launch penalties imposed by fuel required for corrective maneuvers during the outbound leg of the trip should be small.

90/ Table 2 was compiled from the design characteristics of LOX/LH lunar vehicles given in the Northrop, Lockheed, North American Aviation and General Dynamics reports. Because of large uncertainties in the characteristics of future lunar propulsion systems, the results must be considered as approximate and subject to verification. The results were computed on the basis of the following assumptions. (a) the extra fuel needed for velocity corrections is to be used in available propulsion engines (3rd stage for direct flyback missions; lunar takeoff engine in lunar landing missions, etc.) so that the corrections do not involve extra hardware or structual requirements. (b) the corrections

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are applied to a 15000 lb reentry system, that is, all unneeded hardware is jettisoned prior to initiating the correction. (c) 9% structural facor. The results thus must be considered as representing optimistic conditions and may have to be revised upward.

TABLE 2

TAKEOFF PENALTIES IMPOSED BY MIDCOURSE CORRECTIVE MANEUVERS

MISSION PROFILE						
Launch	Moon	Reentry	Characteristic Velocity (fps)	Nominal Launch Weight * (lb)	Extra Launch Weight per Extra lb * (lb)	Extra Launch Weight for 100 fps Correction on a 20,000 lb Vehicle (lb)
Orbital	Circum-lunar	Ballistic	11,000	75 K	5	850
Orbital	Circum-lunar	Retrothrust	23,000	160 K	10	—
Surface	Circum-lunar	Ballistic	36,000	520 K	35	6,000
Orbital	Orbit	Ballistic	17,000	100 K	7	1,150
Orbital	Orbit	Retrothrust	29,000	300 K	20	—
Surface	Orbit	Ballistic	42,000	1 M	70	11,500
Orbital	Landing	Ballistic	29,000	334 K	20	3,300
Orbital	Landing	Retrothrust	41,000	900 K	60	—
Surface	Landing	Ballistic	54,000	3,6 M	250	40,000
Surface	Landing	Retrothrust	66,000	15 M	1000	—

* Assuming V_e = 4 Km/sec (LOX - LH_2)
Structural factor E = .10
4 stages outbound + (1 stage inbound)

[redacted]

weight of the entire system. No serious difficulties should be experienced in providing for the extra fuel needed for midcourse maneuvers, even when the factors discussed earlier (corrections in the 3 dimensional space, control of time of reentry and the like) are taken into consideration. On the other hand, the penalties imposed by midcourse corrections are large when referred to the weight of the "useful" payload (the Apollo capsule). The next to the last line in Table 2 shows that the usable payload may be more than doubled for each reduction of 100 fps in midcourse maneuvers. If and when frequent exchanges between earth and moon become necessary, it will be essential, from an economic standpoint, to minimize midcourse fuel requirements.

2.45 The fourth area requring investigation concerns the optimum scheduling of midcourse corrections. For example, should several small thrusts or a single large one be used to correct trajectory errors? At what point or points should the maneuver(s) be initiated? The answer can be easily derived 91-95/ in the case of a vehicle moving about a single center of attraction, and when either the aim point or the speed at this point is to be corrected. The general problem, involving two center of attractions and control of both space and time coordinates 96/ of the point of arrival cannot be solved analytically.

2.46 In most of the investigations 97-100/ concerned with this problem, the number and time of application of the corrective thrusts are fixed

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91/ J. V. Breakwell, The Spacing of Corrective Thrust in Interplanetary Navigation, Am. Astron. Soc. Meeting, Seattle, 1960.

92/ D. F. Lawden, "Minimal Rocket Trajectories," J. Am. Rocket Soc., 360, p. 23., 1953.

93/ "Fundamentals of Space Navigation," J. Brit. Interplan. Soc. 13, p. 87, 1956.

94/ NASA TR R-80, op. cit.

95/ J. Lorell, "Velocity Increment Required to Reduce Target Miss in Coast Trajectories," Advances in Astronautical Sciences, Vol. 6, 1960.

96/ Control of the reentry speed does not appear to be required for Apollo type vehicles.

97/ NASA Industry Apollo Technical Conf., op. cit.

98/ S. F. Schmidt, et. al., A Study of a System for Midcourse Navigation, NASA Industry Apollo Technical Conf., Washington, D. C. 1961.

99/ J. A. White, A Study of the Guidance of a Space Vehicle Returning to a Braking Ellipse About the Earth, NASA TN D-191, 1960.

100/ D. P. Harry, et. al., Exploratory Statistical Analysis of Planet Approach, NASA TN D 268, 1960.

[redacted]

[redacted]

arbitrarily. Only a few papers[101-103] investigate the scheduling in respect to the optimum utilization of the fuel. In the case of earth orbital launching (Table 2), midcourse corrections impose small relative penalties at take off. Investigation of the optimum in correction scheduling thus does not appear warranted. On the other hand, the heavy take off penalties in surface launch missions (Nova) suggests that investigation of the correction schedule may have a practical interest.

2.47 The fifth area of investigation concerns the accuracy with which the midcourse corrections can be applied in space. There is little point in being able to accurately determine the required midcourse maneuvers if the corrective thrust cannot be accurately delivered.

2.48 Only a few attempts have been made to estimate this accuracy. This uncertainty is expected to vanish as our experience with space probes increases and as the design criteria of the lunar vehicle becomes known.

2.49 Efforts should be made to estimate realistically the expected accuracy in controlling the thrusts, with the purpose of determining the effect of errors on the magnitude of midcourse corrections and thus, on the weight penalties at takeoff.

SUMMARY

2.50 The problem of guiding an inbound lunar vehicle to insure, not only safe reentry within the atmosphere but reliable recovery on the ground has not been adequately covered. An investigation is needed to determine the magnitude of midcourse maneuvers compatible with the reliability level assigned to the mission. The results should be given in terms of the various types of lunar missions considered, in terms of probable launch errors from the earth and/or the moon and should be translated in terms of extra fuel, guidance, computation requirements weights that is, in terms of payload penalties at launch, as well as in terms of the landing area within which rescue facilities are to be disposed.

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[101] D. F. Lawden, Optimal Program for Correctional Maneuvers, Rad. Inc. TR RR 1186-60-13, 1960.

[102] J. A. White, A Study of the Effects of Errors in Measurement of Velocity and Flight Path Angle on the Guidance of a Space Vehicle Approaching the Earth, NASA TN D 957, 1961.

[103] NASA TR R 102, 1962, *op. cit.*

[redacted]

[redacted]

2.51 Some of the questions to be answered are:

a. What is the effect of lunar launch errors on the perigee altitude, the time of arrival, the speed and direction of the vehicle at reentry within the earth's atmosphere?

b. How is the landing area on earth affected by these errors, given a reentry vehicle with specified aeroballistic capabilities?

c. What is the magnitude of velocity corrections corresponding to a predetermined landing area and what are the fuel requirements associated with these corrections, in terms of mission reliability, precision with which corrective impulses can be delivered, etc.

d. How are these fuel requirements reflected in the take-off payloads, in terms of mission profile? Will these penalties affect the validity of present concepts in lunar mission profiles?

e. If the fuel penalties, as well as the penalties associated with inboard navigation, guidance, computing and control facilities prove to be prohibitive, how can the initial specification of lunar missions be relaxed? For instance, will earth-based tracking radar be capable of acquiring the incoming vehicle,[104] and predicitng the location of its foot print and, if so, will it be possible to deploy recovery facilities at suitable locations within an adequate time?

2.52 Some of the programs proposed by NASA may, in time, provide answers to these questions, provided that their scope is enlarged to encompass the over-all concept of lunar mission, rather than being restricted to one particular phase. These relevant programs include:

a. Performance and Guidance Trajectory Studies, proposed by the Aeroballistic Laboratory, MSFC; the scope of this program is, however, directed toward low thrust propulsion systems and may not be applicable to the Saturn or Nova missions.

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[104] Michaels, et. al., Lunik III Trajectory Predictions. Annual Meeting of American Astronautical Society, 1960. It should be noted, incidentally, that trajectory computations for Lunik III on IBM 704 required up to 15 hours for each integration step. Such lag is clearly unacceptable in initiating corrective maneuvers. The problem is further complicated by the "zone of silence" during atmospheric reentry.

[redacted]

b. Orbital Launch Guidance Systems Studies, proposed by the Advanced Studies Branch, Guidance and Control Div. MSFC.

c. Reentry and Return Guidance Studies, proposed by the Advanced Study Branch, MSFC.

d. Feasibility Study of Saturn Real Time Evaluation, proposed by Aeroballistic Laboratory MSFC. The scope of this program is restricted to orbital operations.

ASTRONAUTICS PROJECT A-1

CONTROL OF TIME OF REENTRY

1. Task Statement. To investigate procedures for controlling the time of reentry of an inbound lunar vehicle, with the purpose of maximizing the probability of recovery.

2. Justification. Because of the rotation of the earth, it is essential to control the time of reentry (coordinate with earth rotation) in order to hold to a minimum the area within which the capsule may land. Fuel expenditures for midcourse maneuvers required to control time of reentry must be evaluated.

3. Present Status. Present investigations are limited to control of space coordinates of reentry point to insure safe aerodynamic reentry.

4. Criticality. The control of time of reentry critically affects the disposition and efficiency of recovery air, sea and land facilities. It could affect the transearth midcourse maneuver fuel expenditure (final stage) substantially—and additional fuel and weight will affect first stage design coordination. Thus, this project should be undertaken early in the program.

5. Mission Applicability. Recovery of all manned lunar vehicles:

    a. Earth reentry and land.

6. Reference. Analysis of Astronautics, paragraph 2.9, page 6.

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ASTRONAUTICS PROJECT A-2

COMPUTATION OF THREE DIMENSIONAL TRAJECTORIES

1. Task Statement. To determine the computing requirements imposed by three dimensional midcourse maneuvers.

2. Justification. The trajectory plane of an earth bound vehicle determines the possible landing areas. Errors in earth or lunar launch may require that dog leg maneuvers be initiated to alter the trajectory plane. Three dimensional maneuvers will require inboard computation of capabilities exceeding those required for two dimensional corrections, particularly when the additional problem of also controlling the time of reentry is introduced.

3. Present Status. A few investigations have considered the theoretical aspects of three dimensional navigation and corrections. These results must be extended to the control of time of reentry and translated into terms of computational requirements.

4. Criticality. Required to determine design criteria of inboard computers; thus it should be conducted prior to computer design.

5. Mission Applicability. Early circumlunar mission where small earth launch errors have a large effect on the reentry plane:

   a. Orbital launch and translunar flight.

   b. Lunar launch and transearth flight.

6. Reference. Analysis of Astronautics, paragraph 2.35, page 22.

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ASTRONAUTICS PROJECT A-3

EVALUATION OF LUNAR LAUNCH ACCURACY

1. Task Statement. To analyze expected lunar launch accuracy with the purpose of determining the magnitude of midcourse guidance required to insure safe reentry and recovery.

2. Justification. Future missions will involve lunar surface or orbital launches. It is unrealistic to assume that guidance accuracies obtained in earth launches will apply in the unfavorable environment and isolation of the moon. Expected launch errors should be realistically evaluated to determine midcourse guidance requirements.

3. Present Status. None.

4. Criticality. Applicable data, such as moon atmosphere and crust composition, for this investigation may be obtained from Surveyor missions.

5. Mission Applicability. Lunar landing or orbiting missions:

    a. Lunar orbit and land.

    b. Lunar launch and transearth flight.

6. References. Analysis of Astronautics, paragraph 2.41, page 24.

ASTRONAUTICS PROJECT A-4

INVESTIGATION OF RECOVERY PROBLEMS

1. Task Statement. To determine mission profile optimizing the probability of capsule recovery.

2. Justification. The probability of locating and recovering the capsule after landing depends on the area within which the landing point is expected to be located. Tighter control of the landing area requires higher accuracy in controlling reentry parameters, that is, increased fuel penalties for midcourse corrections. Conditions maximizing the expectancy of success should be determined with the view of defining the disposition of recovery facilities.

3. Present Status. Investigation of this problem is limited to the recovery of orbiting satellites.

4. Criticality. This investigation may point out improved concepts and procedures for recovery of capsule, and improve the over-all reliability of the manned lunar mission.

5. Mission Applicability. Recovery of mannned lunar vehicles:

   a. Earth reentry and land.

6. Reference. Analysis of Astronautics, paragraph 2.15, page 14.

ASTRONAUTICS PROJECT A-5

EARTHBOUND SPACE DYNAMICS

1. Task Statement. To determine optimal corrective maneuvers during the transearth portion of a lunar mission, with the purpose of minimizing fuel expenditure and insuring landing within a specified area.

2. Justification. Fuel expenditures for corrective maneuvers on the transearth flight of an orbital lunar mission impose severe earth launch weight penalties. The magnitude of corrective thrusts decreases, but the navigational errors increase for increasing distances from vehicle to earth. The scheduling (number, time of application, direction) of corrective thrusts minimizing fuel expenditure should be determined as a function of lunar launch errors and the confidence level with which the vehicle is to land within a specified area.

3. Present Status. A few areas of the problem have been treated but there remains a requirement for consolidating the results in terms of the command-service modules contemplated by NASA.

4. Criticality. Minimal fuel requirements must be determined to specify the design criteria of the service module.

5. Mission Applicability. Orbiting and landing lunar missions:

   a. Lunar orbit and landing.

6. Reference. Analysis of Astronautics, paragraph 2.17, page 14.

ASTRONAUTICS PROJECT A-6

SPACE TACHOMETER

1. Task Statement. To develop means for directly measuring the velocity of a vehicle in space.

2. Justification. Velocity is to be derived from observation of the space coordinates of the vehicle at different times. The accuracy with which velocity is obtained thus depend on navigational errors which themselves depend on the location of the vehicle relative to the earth or moon. The velocity accuracy also increases with the interval separating observations. Under some conditions, the delay in obtaining the velocity may be undesirable. Navigation would be improved by the development of an absolute space tachometer.

3. Present State. Doppler radar is practical only at low altitudes above earth or moon. Schemes based on measurements of Doppler shift of Lyman; differential radiation pressure; electron transit time between two points on the vehicle; magnetohydrodynamic interactions with the magnetic field of space; space charge effects have been proposed.

4. Criticality. Not immediate.

5. Mission Applicability. Space travel:

   a. Earth orbital launch and translunar flight.

   b. Lunar launch and transearth flight.

6. Reference. Analysis of Astronautics, paragraph 2.27, page 20.

ASTRONAUTICS PROJECT A-7

LAUNCH PENALTIES IMPOSED BY MIDCOURSE CORRECTIONS

1. Task Statement. To compute launch penalties caused by the weight of fuel required for midcourse corrective thrusts.

2. Justification. The launch penalties depend on the magnitude of characteristic velocity increments imposed by midcourse guidance and on the mission profile. The penalties are highest in a lunar landing mission launched from the earth surface (NOVA). The magnitude of these launch penalties must be realistically evaluated.

3. Present State. The magnitude of corrective thrusts has been evaluated in a few simple cases. Current NASA studies are expected to supply generalized results; this study would translate characteristic velocities in terms of payload increments.

4. Criticality. Probably important in NOVA missions; should be conducted early in development of launch vehicles.

5. Mission Applicability. May indicate preferred types of missions and/or needs for improved midcourse correction procedures:

   a. Earth orbital launch and translunar flight.

   b. Lunar launch and transearth flight.

6. Reference. Analysis of Astronautics, paragraph 2.42, page 25.

[redacted]

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF ASTRONAUTICS

	Project Title	Funding	Origin (MSFC Laboratory)	
1.	Orbital Transfer and Guidance Studies	FY61/$200K	AERO	Restricted to orbital operations.
2.	Performance and Guidance Trajectory Studies	FY62/$120K	AERO	
3.	Analytical Study of Satellite Rendezvous	FY61/$ 50K	AERO	
4.	Ascent Guidance Studies	FY62/$200K	Adv. St. Brch.	Chiefly concerned with control and hardware.
5.	Orbital Launch Guidance Studies	FY62/$ 50K	Adv. St. Brch.	
6.	Reentry and Return Guidance Studies	FY62/$100K	Adv. St. Brch.	
7.	Rigid Body Motions in Space	FY62/$ 50K	RPD	Mostly concerned with attitude control.

General Remarks: Only program No. 2 considers some of the basic navigational and midcourse maneuvers problems discussed in the Astronautics Analysis. The problems of reentry control to insure landing within a specific area are not considered. It is highly desirable to greatly increase NASA efforts within the general area of space navigation and midcourse guidance.

[redacted]

III. ENVIRONMENTAL EFFECTS

3.1 This analysis considers the environmental problems which may be encountered in lunar missions. Three problem areas will be considered, characterized by:

a. the nature of the interplanetary space through which the vehicle is traveling

b. the nature of the lunar surface, on which the vehicle may land

c. the mechanical and electrical effects caused by the motion of the vehicle itself.

The effects of environment of conditions on the crew do not fall within the scope of this investigation and are not considered here. Discussion of the radiation effects arising from the presence of a nuclear propulsion system will be postponed until the characteristics of these systems become available.

3.2 Because of the wealth of reported data, only the most significant conclusions will be presented here, with the purpose of defining the areas requiring further examination. No attempt will be made to present a full bibliography on the effects of space environment since the latter is already available in several reference materials.[1,2,3]

1/ Space and Aeronautics R & D Handbook, Vol. 4 State of the Art.

2/ Satellite Environment Handbook, LMSD 89006, 1960.

3/ L. D. Jaffe, et al., Behavior of Material in Space, JPL N103600 and M. Neugebauer, The Space Environment, JPL TR34-229, 1960.

3.3 Environmental conditions are determined by the characteristics of vehicle and mission. The content of this analysis is limited to the general description of the conditions under which a lunar vehicle may be called to perform. Detailed analysis of environmental problems in terms of mission will be presented in a later section.

Space Environment

3.4 Effects of High Vacuum. Neither the properties of structural materials nor the performance of electronic components is affected by vacuum.[4,5] The slow evaporation of thin films or wires[6] is not expected to present significant problems in short duration lunar trips, or even in the operation of semi-permanent (a few years) space orbital platforms.

3.5 The only potential difficulties to be expected from operation in the high vacuum of space concern:

a. The cold welding of mechanical joints and articulations (periscope, antennas, motor bearings, etc.). In the earth atmosphere, bearing surfaces are protected by a continuously renewed adsorbed film of oxide which helps prevent seizure. No such surface exists in high vacuum. The difficulties are magnified by the high temperatures which may be caused by solar heating or by radiation from the exhaust jet. Incidence of cold welding in vacuum may perhaps be prevented by lubrication with low vapor pressure oils or grease or by the use of low friction surfaces (nylon), or by solid boundary lubrication films[7] (sulfides, etc.). Although these techniques may adequate in short duration missions, the expected increasing complexity and duration of lunar missions may call for lubricants of improved stability or for means of improving the frictional behaviour of surfaces.

4/ J. H. Atkins, Effects of Space Environment on Materials, WADD TR60-721, 1960.

5/ RF-920, Ohio State University, Research Foundation.

6/ R. A. Ladd, Survey of Material Problems Resulting from Low Pressure and Radiation Environment in Space, NASA TN D477, 1961.

7/ WADD TR60-721, op. cit.

A number of investigations along these lines will shortly be initiated by NASA.[8] However, present laboratory techniques (10^{-8} mmHg) cannot reproduce the high vacuum in space (10^{-11} mmHg). There are indications[9] that monolayer adsorption in ultrahigh vacuum is very sensitive to pressure. For these reasons, the applicability of these programs to the development of techniques for operation in space cannot be evaluated at this time.

b. The loss of plasticizer from plastics, resulting in brittleness and the gas evolution within organic materials, subject to ionizing radiation.[10,11] Gas evolution may result in severe degradation in the optical properties of, say, molded plastic lens, in the opacity of Canadian Balsam used to cement lenses in optical instruments, in the extrusion of the potting compound (asphalt) from transformers or similar electronic components. Again, it is believed that such effects will not be appreciable in short duration missions. In long range orbital operation, it will be necessary to develop substitutes for the materials affected by ionizing radiations. Investigations along these lines are pursued at several laboratories[12] and the results show that these difficulties may well be solved within a near future.

8/ Research on Bearing Materials for use in Highland Ultrahigh Vacuum. Development of Solid Film Lubricants. Research on Bearing Materials for Use in Space Environment. Development of Inorganic Polymers for Use in Sealant and Lubricants at High Temperatures and Subatmospheric Pressures. Investigation on the Combed Effects of Nuclear Radiations, Vacuum and Cryogenic Temperatures on Engineering Materials.

9/ R. A. Roche, "The Importance of High Vacuum in Space Environment Simulation," Vistas in Astronautics, Vol. 2, 1959.

10/ H. M. Abbott, Effects of Vacuum and UV Radiation on Polymeric Materials, Lockheed SB 61-20, 1961.

11/ NASA TN D477, op. cit.

12/ MIT, Midwest Research Institute, GE, Franklin Institute, Lytton, Ind., Indiana University, Lewis Research Center.

c. The absence of heat transfer by convection from electrical or electronic equipment. Cooling may be insured by conduction through proper design of the component.

3.6 Meteorites. The impact of even a very small meteorite may have catastrophic results.[13,14] Because the speed of meteorites is generally greater than the speed of sound in the material of the vehicle, the kinetic energy can be dissipated only within the region bounded by the shock front whose volume cannot be larger than a few times that of the particle. Temperatures of the order of 10^6 or 10^7 °K are generated from the impact of a meteorite of typical mass and velocity. If this high temperature spike falls on a sensitive element, an electric wire or a contact surface where the heat cannot be quickly dissipated, permanent damage to an essential component may result from the impact of even a small meteorite. If the impact punctures the cabin or a propellant tank, the high energy may set off a disastrous explosion or deflagration of the cabin atmosphere or the propellants.[15,16]

3.7 Within a massive metalic shield, the heat is quickly transferred by conduction to the surrounding regions. The net effects consist in local melting and recrystallization, lattice dislocation or nuclear transformations resulting in the progressive degradation of the structural integrity of the shield. A given thickness of shielding is effective only against meteorites whose kinetic energy is below a critical value.

3.8 It is generally accepted that protection against all sizes of meteorites is neither practical nor economical. On the basis of the observed relative frequency of meteorite sizes, it is possible to estimate the probability that a shielded vehicle will not encounter a meteorite larger than the size provided for by the shield. For example,[17] a thick-

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[13] F. L. Whipple, "Meteorite Risks to Space Vehicles," Proceedings, VII International Astronautical Conference, Barcelona, 1957.

[14] E. T. Benedikt, "Disintegration Barriers to Space Travel," Advances in the Astronautical Sciences, vol. 5, 1960.

[15] F. T. Smith, Meteoric Problems Related to Space Vehicles, Aeronautic System, Inc., vol. 407, 1959.

[16] R. Meyer, Explosive Failure in Pressurized Space Cabins, Manned Space Station Symposium, Los Angeles, 1960.

[17] F. L. Whipple, op. cit.

ness of 1.28 cm of aluminium will protect a vehicle against the impact of meteorites smaller than 3100 μ radius. The probability of encounter of a 3 meter diameter sphere during a 5 day lunar trip with larger meteorites is then found to be about .01. Furthermore, not all of the impacts with the larger bodies may be expected to have disastrous consequences.

3.9 However, these results are based on a meteorite size frequency distribution determined from radar or visual observation of meteors in the earth atmosphere, as well as from erosion and impact data from space probes.[18/] Because of the relative scarcity of the larger meteorites, their actual density in space has been estimated by extrapolating the data obtained on smaller size bodies. The probability of catastrophic collision, obtained in the manner indicated above may thus be questioned. A critical study by RAND[19/] indicates that the probability of puncture may vary by as much as several thousands, on the basis of presently available information.

3.10 Furthermore, the highest velocities which can be imparted to a projectile in the laboratory does not presently exceed 25,000 fps. It has long been known[20-22/] that the phenomenology of impact varies with the speed. Results obtained at 20,000 fps do not corrolate with those obtained at lower speeds. One may therefore question whether present laboratory results[23-25/] at 25,000 fps can be safely extrapolated to meteorite hypervelocities of the order of 100,000 fps.

[18/] Effects of Micrometeorites on Space Vehicles, an Annotated Bibliography, Lockheed SB-61-37.

[19/] R. L. Bjork, et al., Estimated Damage to Space Vehicles by Meteoroids, RAND RM-2332, 1959.

[20/] A. C. Charters, "High Speed Impacts," Scientific American, No.203, p. 128.

[21/] L. Summer, Investigation of High Speed Impacts, NASA TN D96, 1959.

[22/] S. F. Singer, Effect of Meteoric Particles on a Satellite, Maryland University, TR 41, 1956.

[23/] J. L. Summer, Impact Resistance of Vehicle Structures, NASA Industry Apollo Technical Conference, Washington, 1961.

[24/] S. Katz, et al., Penetration of Metal and Lucite by Small Particles, AFCRC TR 57452, 1957.

[25/] M. R. Liaciardello, Structures in Space, WADC TN 59-13, 1959.

3.11 For these reasons, conclusions on the protection of space vehicles against meteorites must be critically reexamined. The present uncertainty can be resolved either by (a) developing techniques for increasing the velocity of projectiles in the laboratory (electric discharge guns) and (b) statistical tabulation of large meteorite impacts on future space probes. The actuality of this problem may be judged by the number of active[26,27] or proposed[28] projects at NASA or in the industry.[29-32]

3.12 To summarize, meteorite impacts present serious problems to lunar missions, both in regard to the hazards involved and to the penalties in shielding weight. Although constant efforts are being expended in this field, we are just beginning to obtain a general picture of the physical processes involved in hypervelocity collisions. This level of effort must be sustained if results are to be on hand at the time lunar missions become operative.

3.13 In addition to impacts with meteorites of finite sizes, the surface of the vehicle is subject to erosion by the very fine cosmic dust present in space.[33,34] This erosion is not significant in short duration lunar missions or even in orbital missions of few years duration. The only potential hazard is the gradual etching of optical surfaces exposed to the dust. It has been estimated that this effect would cause a glass surface

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26/ J. O. Funkhauser, Preliminary Investigation of the Effects of Bumpers to Reduce Projectile Penetration, NASA TN D802, 1961.

27/ E. H. Davidson, Space Debris Hazards Evaluation, NASA TN D1105, 1961.

28/ Physics of Meteoroid Impacts. Investigation of Spectral Emissivity of Metals After Damage by Particle Impacts. Meteoric Particles Shield Criteria. Development of Critical Impact Velocity Data for Saturn Structural Materials.

29/ F. T. Smith, Meteorite Problems Related to Space Vehicles, Aeronautical Systems, Inc. U-407, 1959.

30/ Material in Space Environment, Syracuse Univ., MET 597-596, 1958, Also, Lockheed, Convair, General Dynamics etc.

31/ R. L. Bjork, A Conservative Estimate of the Meteoroid Penetration Flux, RAND P 1913, 1960.

32/ R. A. Gemmel, Criteria for Meteorides Protection, ARS Conference, Santa Barbara, 1960.

33/ C. W. McCraken, et al, Direct Measurement of Interplanetary Dust, NASA TN D-1174, 1962.

34/ S.F. Singer, Effects of Interplanetary Dust on Space Vehicles, 2nd Symposium on Physics of Space, San Antonio, 1958.

to become inutilizable in about 1.7 years. One NASA program[35] is concerned with this hazard.

3.14 Electric and Magnetic Effects. A body in space is subject to the bombardment of protons and electrons which constitute the cosmic plasma. The speed of these particles correspond to a kinetic temperature of 1000 to 10,000° K. Because of their small mass, the speed of the electron is higher than that of the protons. More electrons than protons impinge on the body and the latter acquires a negative charge (relative to the space charge of space). This charge will increase until the electrostatic field reestablishes a balance between the rate of collisions of electrons and protons.

3.15 In a moving vehicle, the front surface will "overtake" more of the slower moving protons than when at rest; the rear surface will be overtaken by less protons than when at rest.[36] A difference of potential will appear between front and rear surfaces if these surfaces are electrically insulated from each other. Estimates of the potential difference range from a few tens to several hundred volts for a vehicle moving at 36,000 fps. The uncertainty is due to our ignorance of the temperature of the plasma. Potential hazards from electrical discharges may be eliminated by electrically connecting all surfaces of the vehicle. Some problems may still arise in antennas which cannot be grounded.

3.16 The electro-magnetic effects produced by the motion of the negatively charged vehicle in the magnetic fields of the earth or space should be insignificant.

3.17 Electro-magnetic Radiations. The lunar vehicle will be immersed in a complex flux of electro-magnetic radiations. Most of these radiations originate from the sun.

3.18 The solar spectrum corresponds approximately[37] to the emission of a blackbody at a temperature of 6000° K. The radiations thus include a large proportion of infra-red and visible radiations, with a small amount of ultraviolet for a total flux of $1.5\ 10^6$ erg cm^{-2} sec^{-1}. The earth atmosphere is a good (albedo: .34) reflector of solar radiations so that a vehicle in the vicinity of the earth will be subjected both to the direct and reflected

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[35] Investigation of Cosmic Dust Damage to Engineering and Electrical Materials. NASA proposed research program.

[36] Realistic vehicle speeds are so much lower than electron speeds that the rate of electrons impingement on front and back may be considered as constant.

[37] In addition, small amounts of x-rays are emitted by the high temperature corona. The x-ray flux amounts only to a few ergs cm^{-2} sec^{-1} and is negligible before the normal component of the sun and before the secondaries produced within the vehicle by particulate radiation.

solar flux. The reflectivity of the moon surface is poor (albedo: .07) and will not appreciably contribute to the flux. In addition, the earth and moon act as low temperature blackbodies (250-275° K).

3.19 These radiations have negligible effects on the material components of the vehicle. The only notable effect,[38] a slow degradation of polymeric substances by ultra violet, can be easily avoided by shielding the sensitive material from the direct or reflected solar flux.

3.20 The most significant effect of electromagnetic radiations on a lunar vehicle is the heat (and pressure) generated on the vehicle surfaces exposed to the flux.[39] The over-all temperature of the vehicle is determined by the balance between the energy received and that radiated into space. The equilibrium temperature depends on the shape of the vehicle, its orientation relative to sun and earth and the absorption coefficient of its surfaces. Theoretical considerations[40] indicate, and satellite observations confirm[41] that normal (300° K) over-all temperatures can easily be obtained within a lunar vehicle illuminated by the sun. However, if the vehicle is attitude stabilized in space, large differences in temperatures may be present between the dark and illuminated surfaces, resulting in the appearance of mechanical stresses, thermoelectric malfunctions, embrittlement of insulating materials and so on. Temperature differences can be smoothed out either by insuring thermal conduction or convection throughout the vehicle or by spinning it slowly so as to expose all surfaces in turn to the radiant flux.[42] Effect of heat sources (electrical equipment) and heat sinks (cryogenic tanks) can be compensated for by adjusting the absorption and radiative properties of the various surfaces.

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[38] A. L. Alexander, Degradation of Polymers by UV Radiations, NRL, 5257, 1959.

[39] F. G. Cunningham, Earth Reflected Solar Radiation Input to Spherical Satellite, NASA TN D1099, 1961.

[40] J. E. Naugle, Temperature Equilibrium of a Space Vehicle, Vistas in Astronautics, vol. 1, 1959.

[41] L. D. Nichols, Effects of Shielding on the Temperature of a Body from Solar Radiation in Space, NASA TN D578.

[42] Minor problems caused by temperature differentials within the vehicle will be examined in the two following investigations proposed by NASA:

Proposed Electrical Contract Research
Low Temperature Dielectric Coastings

3.21 Much effort is planned by NASA[43] in investigating the problems of vehicle heat control and no serious difficulty is expected in controlling the thermal environment of future lunar missions.

3.22 The radiation pressure caused by the impingement of photons on the vehicle surfaces is extremely small and will not affect the gross trajectory of a massive Apollo type vehicle.[44] Potential difficulties might result from torques generated if the center of pressure does not coincide with the center of gravity. This effect is not expected to be serious in a massive vehicle and may be corrected by the altitude control system.

3.23 To summarize, primary electro-magnetic radiations are not expected to generate serious problems in lunar missions.

3.24 <u>Particulate Radiations</u>. Van Allen and Cosmic Radiations: During periods of quiescent solar activity, the particulate radiations which may affect a lunar vehicle are the electrons and protons trapped in the geomagnetic field (Van Allen belt)[45] and the cosmic rays originating in space. All these particles move at very high speeds; upon impact with the vehicle, their kinetic energy is transformed into heat, x rays, γ rays; some of the impacts may be sufficiently severe to produce nuclear fission with the usual secondary emission of neutrons, x rays or β rays, formation of electron pairs and so forth.

3.25 The discussions of these phenomena and the potential hazards involved may be considerably simplified by observing that, in a manned vehicle, man is the most sensitive component to particulate radiations. If protection of the man is insured by proper selection of the mission profile, then the material components will not, ipsofacto, be affected. If protection of the man is insured by shielding,[46] then it should be possible

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43/ Evaluation of Hemispheric and Spectral Emissivity of Selected Materials. Absorbtivity and Emissivity of Materials. Emittance of Metals at High Temperatures. Low Temperature Thermal Emittance Studies. Theoretical Physics of Emissivity Properties of Solids.

44/ R. W. Bryant, <u>The Effect of Solar Radiation Pressure on Motion of Orbiting Satellite</u>, NASA TN D1063, 1961.

45/ A. J. Dessler, <u>Penetrating Radiations</u>, Satellite Environment Handbook, LMSD 89-5006.

46/ Shielding against the high energy cosmic rays is considered impractical. The flux of cosmic ray is, however, so low that its effects on materials are insignificant. In the following, shielding refers to protection against Van Allen radiations.

to place most of the sensitive components within the shield without excessive weight penalties. The following is therefore limited to the discussion of the effects of particulate radiations on components which cannot be shielded, either because of their bulk or because of their function. The conclusions must be interpreted in the light of our present limited knowledge[47,48] of conditions in space.

3.26 The structural materials of a lunar vehicle are only slightly affected by high energy particulate radiations.[49] Lattice dislocations and possible nuclear transformations from the impact of cosmic rays will progressively cause the structure to become brittle and lose some of the mechanical properties. These effects, however, are negligible in a 10 day lunar mission or even in orbital operations of a few years duration.

3.27 Solar cells, transistors and, more generally all semiconductors are affected[50-52] by high energy electron and proton impacts. Solar cells, of course, cannot be shielded without screening out the solar electro-magnetic radiations. Degradation is slow: failure for solar cells is estimated to result from continuous exposure of a year to the most intense flux in the Van Allen belt, failure of transistors, to exposures of about 8 months.[53] Damage certainly would be negligible during the some 30 minutes required for the passage of an Apollo lunar vehicle through the Van Allen belts or during several years operations at orbital altitude (300 miles) around the earth.

3.28 Effects of particulate radiations on electrical and electronic equipment through metallic sputtering on motor commutators, printed circuits and the like or through ionization of the air between contacts is expected to be

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[47] F. Hollis, Composition of Radiations Trapped in the Geomagnetic Field, AFSWC TN 59-15, 1959.

[48] Satellite Environment Handbook, LMSD 89005, 1960.

[49] J. H. Goodwin, Material Vulnerability to Space Radiations, Aerosciences Labs, JM6-34, 1959.

[50] J. M. Denney, Radiation Damage in Satellite Solar Cells Systems, ARS Conference, Santa Monica, 1960.

[51] R. G. Downing, Electron Bombardment of Silicon Solar Cells, ARS Conference, Santa Monica, 1960.

[52] F. M. Smits, "Solar Cells in the Van Allen Belts," J. Brit. Inst. of Radio Engr., No. 22, 1961, p. 161.

[53] NASA TN D477, op. cit.

negligible in short term missions. One may however expect some increase in electrical noise during passage through the Van Allen belts.

3.29 The only significant effects due to normal particulate radiations will be found in organic materials and glass. Plastics will be subjected to chain breakage, cross linkages and radical formation with subsequent embrittlement, discoloration and gas evolution resulting in degradation of the mechanical or optical properties. Electrical insulation may crack, motor brushes will sputter, potting compound may be forced out of transformer cases, etc. Quartz, glass will become yellow. Optical IR components (NaCl) may be expected to become opaque.[54] However, these effects all occur at radiation dosages far exceeding the lethal dose for human. It should not be difficult to protect sensitive materials adequately or to replace them by less sensitive materials. Considerable effort is being made by NASA[55] along this direction.

3.30 Solar Flares: From time to time, the sun ejects streams of high velocity protons and electrons[56,57] (solar flares). These periods of activity are related to the occurance of sun spots. Their duration varies from a few hours to several days. The energy of solar protons and electrons is many times higher[58] than that of the particles in the Van Allen belt. On the other hand, significant solar events occur so rarely (about once a year), that the probability that the lunar vehicle will be subject to a flare is low. It is estimated[59] that this probability does not exceed a few percents for 5 to 10 days missions.[60,61]

54/ Radiation Damage to Electrical Components, ITT Labs TM 854, 1961.

55/ Protons Shielding Experiments. Space Radiation Shielding. Investigation of Radiation Damage in Engineering Materials. Radiation Effects on Guidance and Control Equipment.

56/ E. P. Ney, "Protons from the Sun," Phys. Rev. Lett, Vol. 3, 1959.

57/ Discussion of Solar Protons Events, NASA TN D671, 1961.

58/ Energy of Solar Protons: 30 50 300 Mev, up to 10 Bev electrons; 100 Mov.

59/ T. Foelsche, et al., Space Radiation Hazards, NASA Industry Apollo Conference, Washington, D.C., 1961.

60/ This probability may be expected to be higher from 1967 to 1973, a period of probable solar activity.

61/ A. J. Dessler, op. cit.

3.31 Because of the shielding effect of the atmosphere, the flux and energy spectrum of solar particulate radiation fluxes can be measured only from space probes. Data presently available[62-64] are neither consistent nor complete. They are based on the observation of a limited number of solar events having occurred within the past few years. Furthermore, the near impossibility to reproduce these fluxes in the laboratory makes it difficult to measure the radiation effects on equipment under controlled conditions.

3.32 A simplified understanding of the relative effects of solar flares and other sources of particulate radiations on material components of a space vehicle is obtained by expressing the irradiance of the source and the maximum permissible exposure in terms of radiant energy. The results are presented in the following table which also includes the permissible exposures for man to provide a basis for comparison.

TABLE 3

Effects of Particulate Radiation on Material Components[65,66]

Irradience:	
Cosmic Rays:	10^3 ergs/gr/year
Van Allen Belt (maximum irradience):	10^8 ergs/gr/year
Solar Flare (each event):	10^5 to 10^8 erg/gr
Maximum Permissible Exposure:	
Plastics and Organic Materials:	10^8 to 10^{10} erg/gr
Electronic Components:	
Ceramic Capacitors	10^{11} erg/gr
Semi conductors (solar cells)	10^8 to 10^{10} erg/gr
Dry Cells	10^8 erg/gr
Transformers, Chokes, etc.	10^8 erg/gr
Optical Glasses:	
Discoloration	10^{10} erg/gr
Unfit for Use	10^{12} erg/gr
Man:	
Chronic Exposure	43 erg/gr/week (.5 r/wk)
Acute Exposure	10^4 erg/gr (9150 rem)

62/ K. G. McCraken, et al, "Comparison of Solar Cosmic Rays Injection," J. Geophys. Res. 65, 1960, p. 2673.

63/ K.A. Anderson, et al., "Observation of Low Energy Solar Cosmic Rays from the Flare of 22 August 1958," J. Geophys. Res. 64, 1960, p. 551.

64/ P. Rothwell, et al, "Satellite Observation of Solar Cosmic Rays," Nature, 65, 1960, p. 799.

65/ JM6-34, op. cit.

66/ MET 597-596, op. cit.

3.33 The results of Table 3 are subject to revision as data from more space probes become available. The results refer only to unshielded exposure because the interaction of high energy radiation with the material of the shield generates secondary emission which cannot be interpreted in a simple fashion. The table shows that exposure to a single high intensity solar flare is equivalent, in terms of irradience, to a one year exposure within the most active region of the Van Allen belt and that man is, by far, the most sensitive component in the space vehicle.

3.34 The problems of protecting material equipment are therefore best examined in the light of the measures which must be taken to protect the crew from lethal radiations.

a. Protection of an Apollo type capsule against a single flare would require shield weights of several tons, depending upon the degree of protection required.[67-69] Such penalty appears to be impracticable at this time, but, should this technique be adopted, it should be possible to develop techniques for retracting the sensitive electrical or optical component within the shield with a minimum additional weight of equipment. The table shows, in fact, that solar cells may be irremediably damaged through exposure to a single solar flare.[70]

b. Present concepts in short duration lunar mission call for abort whenever a flare is anticipated. It is clear that a successful abort is contingent upon our ability to forecast a flare sufficiently in advance to insure the safe return of the crew within the protective atmosphere of the earth. The actual time of travel of solar protons from the sun to the earth (20 min to a few hours) is sufficient to do so. Fortunately, forecasting tech-

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67/ J. Abel, Radiation Designs for Lunar Missions, NASA Industry Apollo Technical Conference, Washington, 1961.

68/ D. H. Robey, "Radiation Shielding Requirements for Two Large Solar Flare Protons," Astron. Acta 6, 1960, p. 206.

69/ T. Foelsche, Protection Against Solar Flare Protons, 7th Meeting, Am. Astron. Soc., Dallas, 1961.

70/ T. Foelsche, Space Radiation Hazards, NASA Industry Apollo Technical Conference, Washington, 1961.

niques,[71] based on the observation of sun spots appear to be capable of giving adequate lead time (a few days). If the mission is aborted, return of the capsule to within the protection of the atmosphere disposes, of course, of the problems of equipment protection.

3.35 Manned lunar bases may be protected by locating them underground. In the absence of information on the composition and nature of the lunar soil, it is impossible to define the problems that such underground installation would generate and the depth required for shielding man and equipment against solar flares.

3.36 The remaining problem consists of insuring the protection of material component in unmanned orbital platforms or in unmanned stations on the surface of the moon. Such stations may be found to be essential to insure communication, guidance, etc. in future transfer operations between the earth and a manned underground permanent lunar base. This type of mission, however, does not appear to be within the realm of our capabilities at this time and the discussion of the relevant problems is postponed.

3.37 To summarize, solar flares present serious hazards to a number of sensitive components even in a short lunar mission. Investigation of the problems concerning the protection of these components does not appear to be justified at present, because all measures taken to protect the crew can be extended to the components. However, the problems posed by solar flares will multiply as the lunar mission profile becomes more complex. Broad new technological areas may have to be investigated in the future to solve these problems.

Lunar Environment

3.38 Because of the absence of a lunar atmosphere, the environment on the surface of the moon is not expected to differ essentially from that in space. Equipment will be subjected to the full impact of meteorites, solar electro-magnetic and particulate radiations as well as to cosmic rays (there should be no lunar Van Allen belt because of the absence of lunar magnetic field). Protection may perhaps be achieved by locating equipment and crew quarters underground. Discussion of potential protective techniques must, however, be postponed until information on the composition, density and structural properties of the lunar soil become available.[72]

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[71] K. A. Anderson, Preliminary Study on Prediction Aspects of Solar Cosmic Rays, NASA TN D-700, 1961.

[72] M. Brunschwig, et al., Estimation of Physical Constants of Lunar Surface, University of Michigan, 3544-1-F, 1961.

3.39 Additional problems concerned with lunar environment include: Structure of the Lunar Crust: Minimal structural standards are required for the landing and take-off of a space craft as well as the material support of a man or surface vehicle. Visual, infrared and radar observations of potential lunar landing areas have been variously interpreted. Some authors[73] maintain that maria are filled with loose dust to a depth of many meters; others[74] contend that the thickness of the dust does not exceed a few millimeters; others[75] affirm the dust to be compacted to the consistency of desert sand; still others[76,77] think that the maria level surfaces are composed of solid lava beds. Even the proponents of lava beds cannot agree on the probable structural properties of the surface: is the lava solid as earth lava deposits or have meteorite bombardment and thermal stresses reduced it to the consistency of pumice, unable to support a space craft or its exhaust? Attempts have been made[78] to simulate lunar landing conditions but the results must, of course, await the gathering of factual data.

3.40 Even the nature of "mountains" emerging from the maria is open to questions. What is the scale of terrain irregularities? Is the surface rolling, so that appropriate landing areas can be easily picked out by an approaching astronaut or is the surface so rugged and broken up as to prevent landing or even excursions by a man on foot?

3.41 It is clear that some of these questions must be answered before manned landing can be attempted. Although some freedom in selecting the landing area is left to the astronaut, the limited fuel capacity of first generation Apollo capsules will severly restrict hovering times.[79] A potential collapse of the soil after landing may cause irremediable damage to the vehicle.

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[73] T. Gold, "Dust on the Moon," Vistas in Astronautics, Vol. 2, 1959.

[74] First Interim Report, ITT, 1959.

[75] F.L. Whipple, "On the Lunar Dust Layer," Vistas in Astronautics, Vol 2, 1959.

[76] G.P. Kuiper, "The Exploration of the Moon," Vistas in Astronautics, Vol. 2, 1959.

[77] H.A. Lang, Lunar Instrument Carrier Landing Factors, RAND, RM1725, 1956.

[78] L. E. Stitt, Interaction of Exhaust Jets with Simulated Lunar Surfaces, NASA TN D1095, 1962.

[79] Each minute of hovering time on the moon increases fuel requirements for lunar landing by 3 to 4%: M.A. Faget, Lunar Landing Considerations, NASA Industry Apollo Technical Conference, Washington, 1961.

3.42 The circumlunar or orbital missions initially contemplated by NASA can only be expected to give a picture of the lunar surface more detailed than is presently available. However, no amount of visual or photographic information can furnish data on the structural properties of the soil. Such data can be obtained only from an unmanned instrument package soft landed on the moon. This is the purpose of the "SURVEYOR" program initiated by NASA. Lack of adequate information, however, does not permit to evaluate the relevancy of this program to the Manned Lunar Mission. Alternatively, little effort appears to be made to develop concepts for landing gears allowing the Apollo capsule to land on problem surfaces such as may be expected on the moon. Failure to have such systems on hand when information on lunar soil becomes available may delay the manned lunar program.

3.43 Thermal Problems: Because of the absence of atmosphere, the lunar surface exhibits considerable temperature variations. The daytime temperature on the equator is estimated[80] at 373° K (100° C), the nighttime temperature, at 120° K (-150° C). During daytime, the temperature of a vehicle or a surface base can be maintained at the normal earth value (300°K) by selecting the latitude of the establishment and/or by controlling the absorption of radiation by means of louvers, etc. The severe temperature differentials (up to 250°) between dark and illuminated surfaces may be reduced by proper design of the vehicle, station or individual space suits, to insure proper heat exchanges between surfaces.

3.44 A potential hazard, not considered so far, consists of changes in absorption characteristics of surfaces and, consequently, changes in the internal temperature, caused by deposition of lunar dust stirred by the exhaust of the craft or the motion of a man.

3.45 During the long lunar night, the temperature will fall to some level between the temperature of the lunar surface (120° K) and that of space (5° K), depending on the shape of the object, its orientation in respect to the surface and the characteristics of the skin. The low nocturnal temperatures may be minimized by internal heat generation[81] or by burying the equipment, taking advantage of the supposedly high insulating value of the lunar soil. Under such condition, the equilibrium temperature would not fall below 250° K, the subsurface temperature of the moon.[82]

80/ First Interim Report, ITT, 1959.

81/ In vacuum, the radiative heat losses may be minimized by selecting surfaces with low absorption coefficients (polished silver or aluminum, etc.).

82/ H.C. Urey, Chemistry of the Moon Subsurface, Int. Symp. on Space Flight, Louvecieune, France, 1961.

3.46 To summarize, the problems generated by lunar thermal environment are not expected to be serious. Present state-of-the-art in vehicle and space suit design should be adequate to insure thermal protection of the crew and material in missions of limited duration. Re-examination of the problems will be required when missions of longer duration or permanent lunar bases are contemplated.

Flight Environment

3.47 This section is concerned with the environment conditions generated by the motions of the vehicle in space or within the earth atmosphere. Because the Saturn or NOVA type vehicles are still in the conceptual stage, the following data represent best estimates based on scale model tests and engineering evaluations.

3.48 Accelerations. The accelerations in a manned vehicle must be limited to a level acceptable to the crew.[83/] Presently accepted limits for axial accelerations are:

Sustained acceleration (take-off)	8 to 10 g's
Temporary acceleration (re-entry)	20 g's
Impact acceleration (landing)	40 g's

These limits refer to accelerations in the eyeball-in direction. During the ballistic phase of the lunar trip, the vehicle operates under conditions of zero acceleration.

3.49 Maximum angular accelerations of 15^{o} sec^{-2} are specified[84/] during operation of altitude control systems.

3.50 Aerodynamic effects in the earth atmosphere, as well as operation of boosters are expected to generate strong vibrations within the 100 to 500 cps spectrum. The noise pressure level for Saturn and NOVA vehicles have been estimated[85-87/] to the following values, on the basis of present

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83/ Creer, et al., Influence of Sustained Accelerations on Certain Pilot Performance Capabilities, NASA Industry Apollo Technical Conference, Washington, 1961.

84/ Project Apollo, Statement of Work, Phase A, NASA, 1961.

85/ W.D. Dorland, Noise Characteristics of Saturn Static Tests, NASA TN D611, 1961.

86/ Criticality of Subsystems for Large Launch Vehicles, Lockheed ER 5388, 1961.

87/ S.A. Stevenson, Payload Vibration Data Measured During Five Flights, NASA TN D963, 1962.

state-of-the-art for smaller boosters:

Aerodynamic	140 to 150 db
Boosters	150 to 175 db

These levels refer to vibrations near the base of the boosters. Attenuation of about 15 db is expected to be observed in the vicinity of the payload. Aerodynamic noise will have its peak at lift off, for a duration of some 30 seconds. Booster noise will have its peak 40 to 60 seconds after lift off and will last 1or 2 minutes. Data refers to LOX/LH_2 boosters. Higher noise levels may be expected with solid boosters.

3.51 In addition to the preceding vibration occuring primarily at earth takeoff, there are indications[88,89] that severe oscillatory motions of the Apollo type capsule may be experienced during re-entry into the atmosphere at parabolic velocities. Because the level and frequency of these vibrations is critically affected by the design of the re-entry body, the re-entry parameters and the pilot maneuvers, no reliable data can be given at this time on this type of environment.

3.52 A number of investigations within these areas has been proposed by NASA.[90] The value of these programs may be limited by the difficulties of reproducing in the laboratory the high noise pressure levels which are anticipated in the operation of Saturn of NOVA boosters.

Thermal Environment

3.53 Thermal environment within the vehicle is determined by:

a. The aerodynamic heating during takeoff or landing within the earth atmosphere.

b. The heating from combustion chambers and exhaust.

c. The cryogenic propellants.

3.54 The flight profile of lunar boosters at or immediately after takeoff

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88/ M. T. Moulton, et al., Dynamic Stability and Control Problems of Plotted Re-Entry from Lunar Missions, NASA Industry Apollo Technical Conference, Washington, 1961.

89/ S.C. Sommer, et al., Study of the Oscillatory Motions of Manned Vehicles Entering the Earth Atmosphere, NASA Memo 3-2-59A, 1959.

90/ Research on Reduction of Vibration Data. Research in the Field of Environment Accelerations. General Study of the Motion of Liquids in Containers and Vibratory.

will not appreciably differ from that of conventional rockets.[91,92] Aerodynamic heating problems should therefore fall within present state-of-the-art practice and need not be discussed further here.

3.55 The extent of aerodynamic heating during re-entry depends on the re-entry profile and the characteristics of the vehicle. The re-entry problems have been extensively studied[93-95] and only general conclusions can be presented here. For the Apollo type of manned capsule, the re-entry profile is dictated by the requirement to hold the deceleration to values acceptable to the crew. This is achieved by having the capsule re-enter the atmosphere at very shallow angles. The aerodynamic heating is then reduced to values comparable in magnitude to those experienced by orbital vehicles during re-entry. Solution of aerodynamic re-entry problems in manned lunar missions thus appear to be within present capabilities[96] if the trajectory of the vehicle at re-entry can be suitably controlled. The re-entry guidance problems are discussed in a separate section.

3.56 Technical investigations, proposed by NASA[97] are expected to further consolidate the present state-of-the-art on ballistic re-entry.

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[91] Proposal for Orbital Docking Test Program, Lockheed LMSD 89-5088, 1961.

[92] Large Launch Vehicle System for a Manned Lunar Landing Program, General Dynamics AE61-0967, 1961.

[93] C. Gazley, "Deceleration and Heating of a Body Entering Planetary Atmospheres," Vistas in Astronautics, Vol. 1, 1958.

[94] F.R. Riddel, et al., "Differences Between Satellite and Ballistic Missile Re-Entry Problems, Vistas in Astronautics, Vol. 11, 1958.

[95] NASA Project Apollo Working Paper No. 1023, 1961.

[96] J. Frisch, "A Nomographic Method of Material Selection for Ablating Shell Structures," Proceedings in Advances in Astronautical Science, New York, 1960.

[97] Heat Conduction through an Ablating Surface for Optimum Heat Protection. Development of Powdered and Fiber Refractory Materials in Combination with Ceramics for Ultra High Temperature Applications, Development of Ceramic Fibers for Reinforcement in Composite Materials, Development of High Temperature Inorganic or Semiorganic Film Forming Polymers, Determination of Thermal Properties of Materials at Temperature Range from -250 to 1500° C, Investigation of Thermal Conduction of Non-Metallic Materials.

3.57 No acceptable analytical method is available to quantitatively predict the severity of base heating by convection and radiation of exhaust jet.[98] Early model cluster firing tests will be required to establish the magnitude of the problem and the necessary design criteria.

3.58 A number of theoretical studies,[99] proposed by NASA are expected to speed up the development of thermal protection devices when the actual heat output of large boosters is measured.

3.59 The problems of minimizing the heat losses from cryogenic tanks have not been solved, particularly in the case of low temperature LH_2 tanks. The difficulties[100] consist in fastening the insulating materials to the tank surface and maintaining its integrity under the accelerations and vibrations at takeoff; preventing excessive frost deposits and insuring the operation of electrical or mechanical components in areas of local low temperatures.

3.60 Investigations in these areas have been proposed by NASA[101] and are expected to eliminate some of these difficulties.

<u>Summary</u>

3.61 This analysis discussed briefly the problems associated with the environment in a lunar mission. An attempt has been made to classify in Table 4 the most important of the problems in terms of the substages of manned lunar missions. For obvious reasons, only the barest description of the problems is presented and the table must be considered only as a reference guide to the accompanying text.

98/ <u>Subsystems for Large Launch Vehicles</u>, Lockheed ER5388.

99/ Base Heating Studies, Base Heat Transfer Measurement in Shock Tubes, Determination of Thermal Properties of Materials at Temperature Ranging from −250 to 1500° C.

100/ Large Launch Vehicles for a Manned Lunar Landing Program, General Dynamics AE61-0967, 1961.

101/ Study to Control and Avoid Adverse Frost. Optimization of Pressure in Cryogenic Tanks, Including Transfer, Storage, and Thermal Insulation. Low Temperature Fatigue Properties of Metals and Alloys. Development of Adhesives for Very Low Temperatures. Investigation of Thermal Conductivity of Nonmetallic Materials. Development of Low Temperature Dielectric Coatings for Electrical Conductors.

[redacted]

TABLE 4

ENVIRONMENTAL EFFECTS ON MATERIALS

	EARTH LAUNCH	ORBITAL LAUNCH	BALLISTIC TRANSIT	MOON LANDING	MOON TAKEOFF	LUNAR BASE	EARTH REENTRY	LANDING
MECHANICAL ENVIRONMENT								
Acceleration (g's)	8	10	0 cryogenic feed	.5	.25	.15	up to 20	up to 40
Shocks, vibrations	150 to 175 db effects on components, effects on structures						? structure components	shocks absorbers
Dynamic pressure	800 lb/ft^2 effects on structures						? effects on controls	
Ambient pressure			vacuum operation of mechanical system					
Nature of surface				characteristics of lunar surface unknown needs for landing gears, vehicles, etc.				dev. of landing gears
THERMAL ENVIRONMENT								
Aerodynamic heating	700°F						2000° F ? thermal control	
Base heating	heat flux not known insulation of structure and cryogenics							
Cryogenics	insulation structural fatigue at low T thermal gradients		insulation operation of mechanical and electrical components at low T					
Solar Radiations						temperature control at night		
SPACE ENVIRONMENT								
Meteorites			incidence of large meteorites not adequately known tradeoff between survival probability and shield penalties					
Particulate radiations			potential problems in long duration missions					
Solar flares			incidence of flares not adequately known effects on electrical and optical components					

[redacted]

[redacted]

ENVIRONMENTAL EFFECTS PROJECT EE-1

BASE HEATING STUDY

1. Task Statement. To protect a space vehicle from booster exhaust thermal flux.

2. Justification. Base heating of large NOVA and/or Saturn boosters is expected to be larger than that of conventional propulsion systems. Means must be provided to reduce the convective and radiant heat transfer to structure and cryogenics.

3. Present Status. Neither experimental data nor satisfactory analytical methods are available to predict the severity of base heating with large boosters. The elements of the problems will have to be deteremined through static test when the boosters become available. Two programs have been proposed by NASA:

   a. Base heating studies

   b. Base heat transfer measurements in shock tubes

   In addition, several related programs on development of insulating materials are expected to provide some preliminary information.

4. Criticality. The magnitude of base heating in NOVA or Saturn must be determined as soon as possible in order to prevent design modifications from delaying the program.

5. Mission Applicability. Surface and orbital launch of large boosters:

   a. Earth Launch and Orbit Mission

   b. Earth Orbital Launch Mission

6. References. Analysis of Environmental Effects, paragraph 3.57, page 57.

[redacted]

ENVIRONMENTAL EFFECTS PROJECT EE-2

ULTRA-HIGH VACUUM RESEACH

1. Task Statement. To test the operation of conventional mechanisms in space environment, with the purpose of determining the need for improved techniques.

2. Justification. There are reasons to believe that seizure may occur in mechanical systems under prolonged exposure to the ultra-high vacuum of space, within the range of temperatures experienced in a space vehicle.

   Because space environment cannot be duplicated in the laboratory and because absorption may be critically altered at low pressures, there is a need to test the adequacy of present lubrication techniques in the actual space environment.

3. Present Status. No program for testing in space has been proposed.

4. Criticality. Prerequisite of Environmental Effects Project EE-3.

5. Mission Applicability. To all mechanical linkages (periscope, antennas valving, gimballing, etc.) used in space vehicles and subject to space environment:

   a. All missions.

6. Reference. Analysis of Environmental Effects, paragraph 3.5, page 40.

ENVIRONMENTAL EFFECTS PROJECT EE-3

ULTRA-HIGH VACUUM LUBRICATION

1. Task Statement. To develop improved lubricants and/or techniques for operation of mechanical systems in space vehicles.

2. Justification. There are reasons to believe that seizure may occur in mechanical systems under prolonged exposure to the ultra-high vacuum in space, within the range of temperatures experienced in space vehicles.

   If this is substantiated in Environmental Effects Project EE-2, there is a need for developing improved low vapor pressure lubricants and/or low frictional bearing surfaces.

3. Present Status. Present and proposed programs should provide improved lubrication techniques for early stages of the manned lunar mission.

4. Criticality. Will be determined from the results of Environmental Effects Project EE-2.

5. Applicability. All mechanical linkages (periscopes, antennas, valving bimgalling, etc.) subject to space environment:

   a. All missions.

6. Reference. Analysis of Environmental Effects, paragraph 3.5, page 40.

[redacted]

ENVIRONMENTAL EFFECTS PROJECT EE-4

PROTECTION AGAINST SOLAR FLARES

1. Task Statement. To protect sensitive equipment in unmanned orbital or lunar stations against damage by solar flares.

2. Justification. Permanent unmanned orbital or lunar stations may be required for navigation, transfer or communications in future lunar missions. There is a need to develop techniques for protecting the sensitive components (solar cells, transistors, etc.) in these stations against solar flares.

   Because the high fluxes of high energy particles cannot be duplicated in the laboratory, it will be necessary to test the validity of concepts of protection by actual operation in space probes.

3. Present Status. A number of programs have been proposed by NASA:

   a. Protons Shielding Experiments

   b. Space Radiation Shielding

   c. Investigation of Radiation Damage in Engineering Materials

   d. Radiation Effects on Guidance Control Equipment

   In the absence of more definite information on the scope of these programs, their pertinancy to the problem cannot be evaluated.

4. Criticality. This program may be postponed until more information becomes available on the nature and scope of future generation lunar missions.

5. Applicability. Semi permanent or permanent unmanned orbital satellites, lunar stations, space probes, and other long term missions:

   a. Future generation missions.

6. Reference. Analysis of Environmental Effects, paragraph 3.30, page 49.

[redacted]

ENVIRONMENTAL EFFECTS PROJECT EE-5

NEW CONCEPTS OF PROTECTION AGAINST SOLAR FLARES

1. <u>Task Statement</u>. Develop new concepts and techniques for the protection of essential components in space against the high energy protons from the sun.

2. <u>Justification</u>. Damage by solar flare to even a minor component unmanned communication or orbital launch station cannot be tolerated for economic reasons. Material shielding may be the answer to protecting small components (Project EE-4) but may be found to be impractical.

3. <u>Present Status</u>. Electrostatic shieldings appear to be impractical. Magnetic shielding may perhaps be practical for the protection of some small component. Development and evaluation of new concepts are needed.

4. <u>Criticality</u>. Essential in future lunar missions involving permanent or semi-permanent space stations.

5. <u>Applicability</u>. Eventually, the concept may be extended to the protection of a manned vehicle, eliminating the need for mission abort:

    a. Future generation missions.

6. <u>Reference</u>. Analysis of Environmental Effects, paragraph 3.30, page 49.

ENVIRONMENTAL EFFECTS PROJECT EE-6

PREDICTION OF SOLAR FLARES

1. Task Statement. To develop techniques for forecasting the incidence of solar flares.

2. Justification. Protection against solar flares of sensitive equipment in permanent lunar or space unmanned stations will probably require either interruption of the normal functions of the station (see Project EE-4) or the expenditure of energy (see Project EE-5). In either case, it is essential that the occurance of the flare be predicted to initiate the protective measures with a minimum down time in the station operations.

3. Present Status. K. A. Anderson of NASA has outlined a technique for the forecast of solar flare. This technique must be fully developed and its reliability tested.

4. Criticality. Forecast of solar flares is essential to insure abort of lunar manned mission and continued operation of unmanned stations.

5. Applicability. All manned space missions; all unmanned communication, guidance vehicles in space or on the moon.

6. Reference. Analysis of Environmental Effects, paragraph 3.34, page 51.

ENVIRONMENTAL EFFECTS PROJECT EE-7

MAPPING SPACE AREAS OF HIGH METEOROIDES DENSITY

1. Task Statement. To determine the space distribution of large meteoroides.

2. Justification. Hazards caused by collision of a lunar vehicle with the larger size of meteoroides cannot be discounted. Meteoroide belts having a high population density (Perseides, Leonides, etc.) have long been known. However, there is no reason to assume that size and frequency distributions coincide. A knowledge of the locations along the earth orbit where large size meteoroides are prevalent would permit to minimize hazards by proper selection of the time of launch and mission profile.

3. Present Status. A RAND Report (P-A13, 1913, 1960), has attempted to define the meteoroide flux in terms of mass and velocity, that is, in terms of potential collision hazards. Too few data are available to set forth reliable conclusions.

4. Criticality. The results should be made available before the launching or permanent space satellites or the establishment of surface lunar stations.

5. Applicability. All manned and unmanned space missions.

6. Reference. Analysis of Environmental Effects, paragraph 3.9, page 43.

ENVIRONMENTAL EFFECTS PROJECT EE-8

SHIELDING AGAINST METEORITES

1. Task Statement. To develop techniques for reducing hazards of meteorite impacts on essential components in lunar vehicles and to test these techniques in space probes or satellites.

2. Justification. While meteorite risk is believed to be slight in a few days lunar mission, the hazards may be expected to become serious in orbiting stations or permanent lunar bases. Areas requiring investigation include:

    a. Development of theories on hypervelocity impacts.

    b. Development of hypervelocity test techniques.

    c. Trial of promising shielding materials in space probes.

3. Present Status. See Technological Support Evaluation.

4. Criticality. This area will become critical when permanent space stations are established.

5. Applicability. Permanent manned or unmanned orbital or lunar stations.

6. Reference. Analysis of Environmental Effects, paragraph 3.11, page 44.

ENVIRONMENTAL EFFECTS PROJECT EE-9

MINIMIZING THE CONSEQUENCES OF METEOROIDE IMPACTS

1. Task Statement. To develop procedures to minimize the damage resulting from a meteoroide impact.
2. Justification. All damaging meteoroides impacts need not be catastrophic. In many cases, the crew might be able to initiate countermeasures to minimize the after effects of impact which, if left unattended, might result in the loss of the vehicle. Procedures should be developed, and equipment designed, to cope with hazardous situations which may be encountered.
3. Present Status. None.
4. Criticality. May improve the reliability of the over-all mission.
5. Applicability. All manned lunar missions.
6. Reference. Analysis of Environmental Effects, paragraph 3.6, page 42.

ENVIRONMENTAL EFFECTS PROJECT EE-10

CONTROL OF LUNAR DUST

1. Task Statement. To develop technique for minimizing thermal effects resulting from the deposition of lunar dust on radiating surfaces.

2. Justification. There are indications that fine dust covers the lunar surface. Deposits of dust on the vehicle or space suits of the crew may alter the absorption and emission characteristics of the surfaces and radically affect the heat balance. It is essential to evaluate the magnitude of this effect and develop techniques for either removing the dust from surfaces or for counteracting the thermal effects.

3. Present Status. None.

4. Criticality. Problems must be solved before lunar landing is attempted.

5. Applicability. Lunar landing vehicles, space suits, lunar bases:

    a. Orbit and Lunar Landing.

6. References. Analysis of Environmental Effects, paragraph 3.39, page 53.

ENVIRONMENTAL EFFECTS PROJECT EE-11

LUNAR ALTIMETER

1. Task Statement. To develop devices indicating the altitude of the craft during landing on the moon.

2. Justification. The SURVEYOR program is expected to provide information on the constitution of the lunar surface. Some authors believe that a thick layer of dust covers the surface. If this proves to be the case, the dust, stirred by the exhaust of the craft, may prevent observation of the ground. Radar and echo sounding are, of course, inutilizable. Techniques are needed to show the altitude of the craft during the last phases of landing.

3. Present Status. None.

4. Criticality. Before lunar landing may be attempted.

5. Applicability. Unassisted lunar landing mission:

    a. Lunar Orbit and Landing

6. Reference. Analysis of Environmental Effects, paragraph 3.41, page 53.

ENVIRONMENTAL EFFECTS PROJECT EE-12

LUNAR LANDING GEARS

1. Tast Statement. To develop versatile landing gears for lunar vehicles.
2. Justification. Because of cost considerations, the SURVEYOR program is expected to provide data on characteristics only from a few points on the lunar surface. Because the extent of hovering in early missions will be limited, the manned lunar vehicle will be required to land on and take off from an uncharted location. Versatile landing gears, capable of operation under a variety of adverse conditions must be provided to minimize the possibility of collapse during landing or take off.
3. Present Status. Inexistent.
4. Criticality. This program must be undertaken as soon as data from SURVEYOR becomes available to provide design characteristics for the first manned vehicles.
5. Applicability. Manned lunar vehicles:
   a. Lunar Oribt and Landing
6. Reference. Analysis of Environmental Effects, paragraph 3.42, page 54.

ENVIRONMENTAL EFFECTS PROJECT EE-13

MINIMIZATION OF CRYOGENIC LOSSES

1. Task Statement. To develop new concepts for minimizing cryogenic losses in space missions.
2. Justification. Cryogenic losses during lunar trips are penalized by increased launch weights. Present concepts, based on insulating LOX and LH_2 tanks are beset by difficulties in developing adequate insulation and fastening it to the tanks. It might be possible to take advantage of the inexhaustible heat sink (5° K) of space to maintain the cryogenics to temperature consistent with low rates of evaporation.
3. Present Status. Inexistent.
4. Criticality. Small.
5. Applicability. All space missions.
6. References. Analysis of Environmental Effects, paragraph 3.59, page 57.

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF ENVIRONMENTAL EFFECTS

CATEGORY: HIGH VACUUM LUBRICATION

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Research on Bearing Materials for Use in High and Ultra High Vacuum	FY61/$ 75K	ORDAB	Possible duplication of efforts.
2.	Research on Bearing Materials for Use in Space Environment	FY62/$100K	M-S&M-ME	
3.	Development of Solid Film Lubricants for Space Environment	FY62/$150K	S&M	
4.	Silicon-Nitrogen Polymer Synthesis	FY62/$ 3K	M-P&VE-MG	Continue NAS 8-1510.
5.	Study of Polymers Containing Si-N Bonds	FY62/$ 70K	MS&M-MG	Possible duplication of efforts.
6.	Development of Inorganic Polymers for use as Sealants and Lubricants	FY61/$ 40K	ORDAB	Continue NAS 8-1510.
7.	Research in the Field of Environmental Acceleration	FY62/$100K	M-G&C-R	Worthwhile attempt to speed up evaluation of new lubrication techniques.
8.	Advanced Ultra High Vacuum Pump	FY61/$150K	ORDAB	

<u>General Remarks:</u> Basic considerations of mean free path indicate that the high vacuum in space cannot be obtained by conventional techniques. Program No. 8 thus does not appear just yet. If lubrication is really based on absorption of monolayers, there are serious doubts that experimentation under laboratory vacuum will be applicable to conditions in space. Efforts should be made to extend the above programs to actual measurements in space satellites.

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF ENVIRONMENTAL EFFECTS

CATEGORY: METEOROIDS

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Physics of Meteoroid Impact	FY62/$45K	RPD	New.
2.	Meteorite Particle Shield Criteria	FY61/$55K	RPD	New.
3.	Hyperacceleration of Micro-particles	FY61/$80K	RPD	Continuation of SRI program.
4.	Microparticlar erosion of Vulnerable Surfaces	FY61/$48K	RPD	Continuation of North American Aviation program.
5.	Acoustical Detection of Meteoric Impacts	FY61/$65K	RPD	Detection devices for space vehicles.
6.	Investigation of Explosive Hazards of LH-LOX to Spark and Squib Ignition	FY61/$45K	RPD	New.

<u>General Remarks:</u> From a practical standpoint, the first four programs should have been preceded by the determination of the frequency of occurrence of meteoroids in space, in terms of their mass and velocity. These data are not now adequately known. These data would indicate the pertinency of the proposed programs and would permit the evaluation of a realistic tradeoff between shield penalties and catastrophic collision probability. Program No. 6 should be extended to the evaluation of hazards associated with penetrating impact of meteorites.

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM
FOR TECHNICAL AREA OF ENVIRONMENTAL EFFECTS

CATEGORY: PARTICULATE RADIATIONS

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Radiation Effects on Guidance and Control Equipment	FY62/$ 75K	M-G&C-R	Chiefly concerned with radiation from nuclear reactor.
2.	Investigation of Cosmic Radiation Damage to Engineering Models	FY61/$ 95K	ORDAB	
3.	Space Radiation Shielding	FY62/$ 50K	RPD	Continue ORD 832.
4.	Protons Shielding Experiments	FY62/$150K	RPD	Theoretical study which may provide new concepts of protection.

<u>General Remarks:</u> Shielding, at best, is practical only against Van Allen radiations which are not expected to present problems in early lunar missions. .
Shielding against cosmic rays and solar flares appears impractical—investigations are needed to develop new comcepts of protection and solar flare prediction techniques.

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EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF ENVIRONMENTAL EFFECTS

CATEGORY: VEHICLE TEMPERATURE CONTROL

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Theoretical Physics of Emissive Properties of Solids	FY61/$35K	RPD	
2.	Evaluation of Hemispherical and Spectral Emissivity of Selected Materials	FY61/$40K	RPD	
3.	Heat Transfer Measuring Techniques	?		Possible duplication.
4.	Pigments Particle Size Investigations	FY61/$20K FY62/$10K	ORDAB-DSNF M-S&M-MG	
5.	Passive Temperature Control	FY62/$ 3K	M-S&M-ME	
6.	Low Temperature Thermal Emittance Studies	FY62/$45K	RPD	Continuation of ADL program.
7.	Investigation of Spectral Emissivity of Metals after Damage by Particulate Impact	FY61/$50K	RPD	Possible duplication.
8.	Microparticular Erosion of Vehicle Surfaces	FY61/$48K	RPD	

General Remarks: The criticality of the first five programs is not established in the light of present knowledge and state-of-the-art.

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IV. GUIDANCE

INTRODUCTION

4.1 Guidance of a launch vehicle is the gathering and analysis of intelligence, and the subsequent decisioning with which to maneuver the vehicle along the flight path and course required to reach a specified destination, at a given velocity and at a given time. It is the culmination of the coordinated functioning of most of the vehicle systems; among these are the guidance, tracking, communication, and control systems, and their subsystems.

4.2 Thus, the interdependence of the Guidance Technical Area and other technical areas is evident. With this interdependence of technical areas, tradeoff considerations are numerous and interface problems are significant. Mission accuracy versus correction requirements are trade-off considerations associated with every mission and its functions. The advantages and disadvantages of each and the capabilities in both areas will be analyzed in establishing specific requirements.

4.3 The Guidance Technical Area includes considerations relative to: receipt of necessary information from various sensors within and external to the launch vehicle guidance system, the conversion of this information to an appropriate form, the computation and analysis of these inputs, the resulting decisions, the conversion of the decisions to commands, and the forwarding of commands to the response centers. Figure 4 is a typical guidance block and information flow diagram of a launch vehicle.

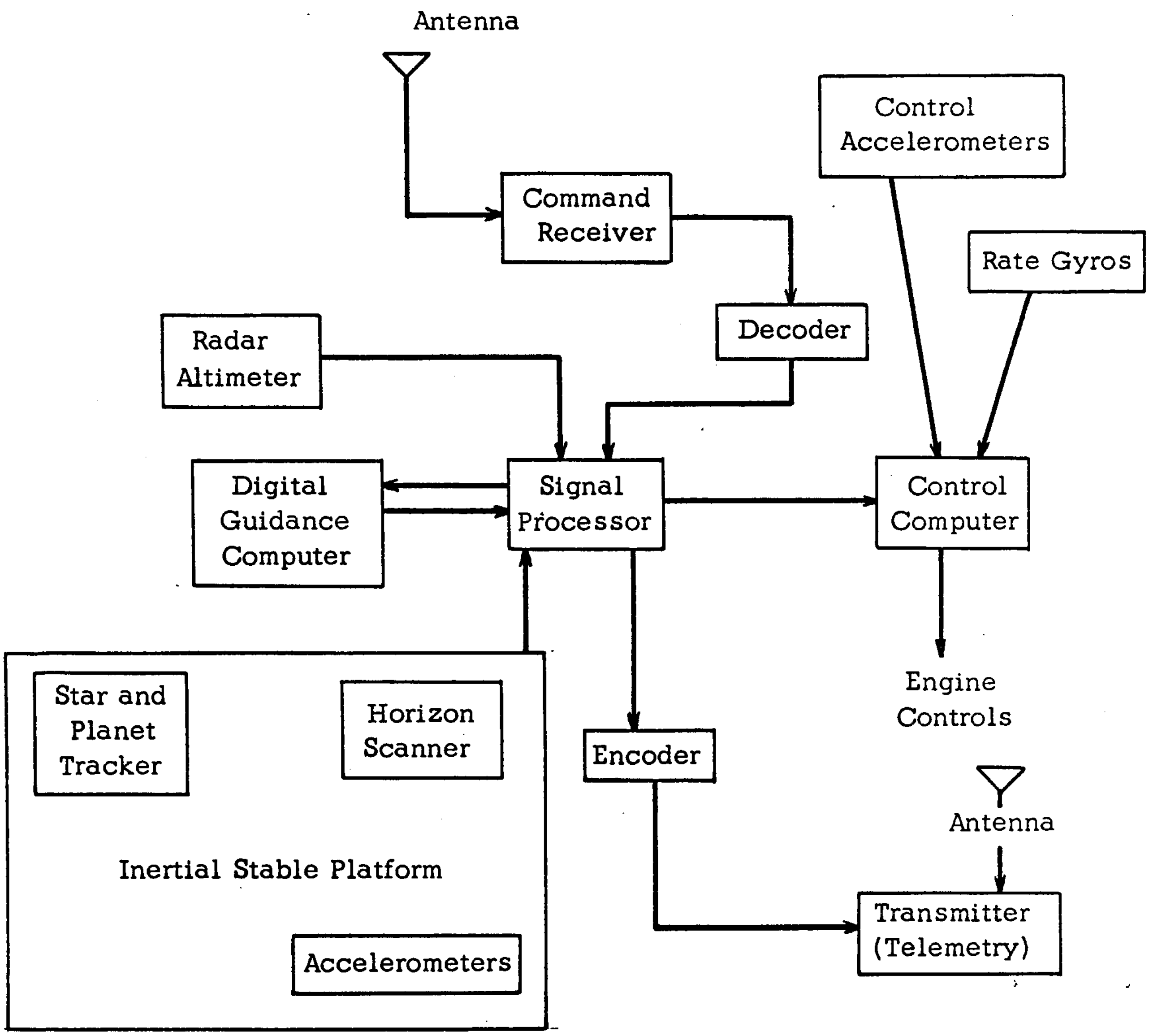

FIGURE 4. GUIDANCE BLOCK AND INFORMATION FLOW DIAGRAM

4.4 Mission parameters establish guidance requirements. For a program as broad as the Manned Lunar Program which includes a number of missions, each capable of being accomplished by more than one method or technique, the parameters become overwhelming in scope and number. However, a discussion will be made of parameters and the resulting guidance requirements based on pessimistic criteria of the various missions and probable methods of accomplishment.

4.5 This study only concerns the guidance of what may be considered first generation vehicles and their probable missions. It will not include guidance requirements of subsequent generation vehicles, that is, the vehicles capable of (1) accomplishing futuristic missions not considered in this study, and (2) undertaking missions considered within the scope of this study, but by utilizing techniques well beyond current and projected state-of-the-arts of the time period under question. However, research and study effort directed toward these subsequent generation vehicles will be discussed.

4.6 Table 5 lists the events during which guidance occurs in the various manned lunar missions for direct flight and interrupted flight modes. Actually, this represents only a portion of possible types of flights derived from the flight sequence diagram in Figure 5. Since most of the Manned Lunar Mission effort is still in the planning and research phases, decisions as to the flight modes are being held in abeyance. Therefore, guidance considerations will reflect the requirements of the most probable flight types resulting from the various sequence combinations feasible over the time period under consideration outlined in Figure 6 and the Apollo mission sequence outlined in Table 6.

4.7 Guidance requirements as well as the (state-of-the-arts) associated with the various manned lunar launch vehicle missions will be discussed in subsequent paragraphs. Guidance performances described in these sections are required during and following the expected environments outlined in the Environment Technical Area Plan.

4.8 In outlining guidance requirements and capabilities, magnitudes and tolerances presented herein include allowances for errors of measurement and errors due to response and functioning of the guidance systems.

4.9 Guidance of vehicles, especially of first generation vehicles, may involve ground support techniques. Although ground support capabilities are not considered within the scope of this study, guidance requirements and capabilities utilizing these techniques will be discussed. It is too closely associated with vehicle guidance of first generation vehicles to be able to discriminate between non-ground supported and ground supported

TABLE 5

PRIMARY MISSION/SECONDARY MISSION/EVENT

EVENTS	SECONDARY MISSIONS	PRIMARY MISSIONS: Cislunar Flight		Circumlunar Flight		Lunar Orbit		Lunar Land	
		Direct	Interrupted	Direct	Interrupted	Direct	Interrupted	Direct	Interrupted
Launch Pad Checkout	Earth Launch & Orbit	X	X	X	X	X	X	X	X
Earth Launch		X	X	X	X	X	X	X	X
Earth Ascent		X	X	X	X	X	X	X	X
Parking Orbit Injection			X		X		X		X
Parking Orbit Sustenance			X		X		X		X
Orbit Transfer			X		X		X		X
Earth Orbit Injection			X		X		X		X
Earth Orbit Sustenance			X		X		X		X
Outbound Orbital Rendezvous	Orbital Rendezvous		X		X		X		X
Outbound Orbital Docking	Orbital Docking		X		X		X		X
Post Dock Checkout			X		X		X		X
Material Pre-Transfer Checkout	Orbital Transfer, Assembly, Repair, Maintenance & Checkout.		X		X		X		X
Material Transfer			X		X		X		X
Pre-Orbital Launch Checkout			X		X		X		X
Earth Orbital Launch			X		X		X		X
Outbound Translunar Trajectory Injection	Earth Orbital Launch & Translunar Flight	X	X	X	X	X	X	X	X
Outbound Translunar Trajectory Sustenance		X	X	X	X	X	X	X	X
Outbound Midcourse Correction		X	X	X	X	X	X	X	X
Outbound System Checkout		X	X	X	X	X	X	X	X
Lunar Orbit Injection	Lunar Landing.					X	X	X	X
Lunar Orbit Sustenance						X	X	X	X
Lunar Descent								X	X
Lunar Hover or Land								X	X
Lunar Pre-Launch Checkout	Lunar Launch & Transearth Flight							X	X
Lunar Launch								X	X
Lunar Ascent								X	X
Lunar Orbit Injection									X
Lunar Orbit Sustenance									X
Lunar Orbital Launch						X	X		X
Transearth Trajectory Injection			X		X	X	X	X	X
Transearth Trajectory Sustenance		X	X	X	X	X	X	X	X
Transearth Midcourse Corrections		X	X	X	X	X	X	X	X
Transearth System Checkout		X	X	X	X	X	X	X	X
Earth Orbit Injection	Earth Re-entry & Land		X		X		X		X
Earth Orbit Sustenance			X		X		X		X
Inbound Rendezvous									X
Inbound Docking									X
Earth Re-entry		X	X	X	X	X	X	X	X
Earth Land		X	X	X	X	X	X	X	X

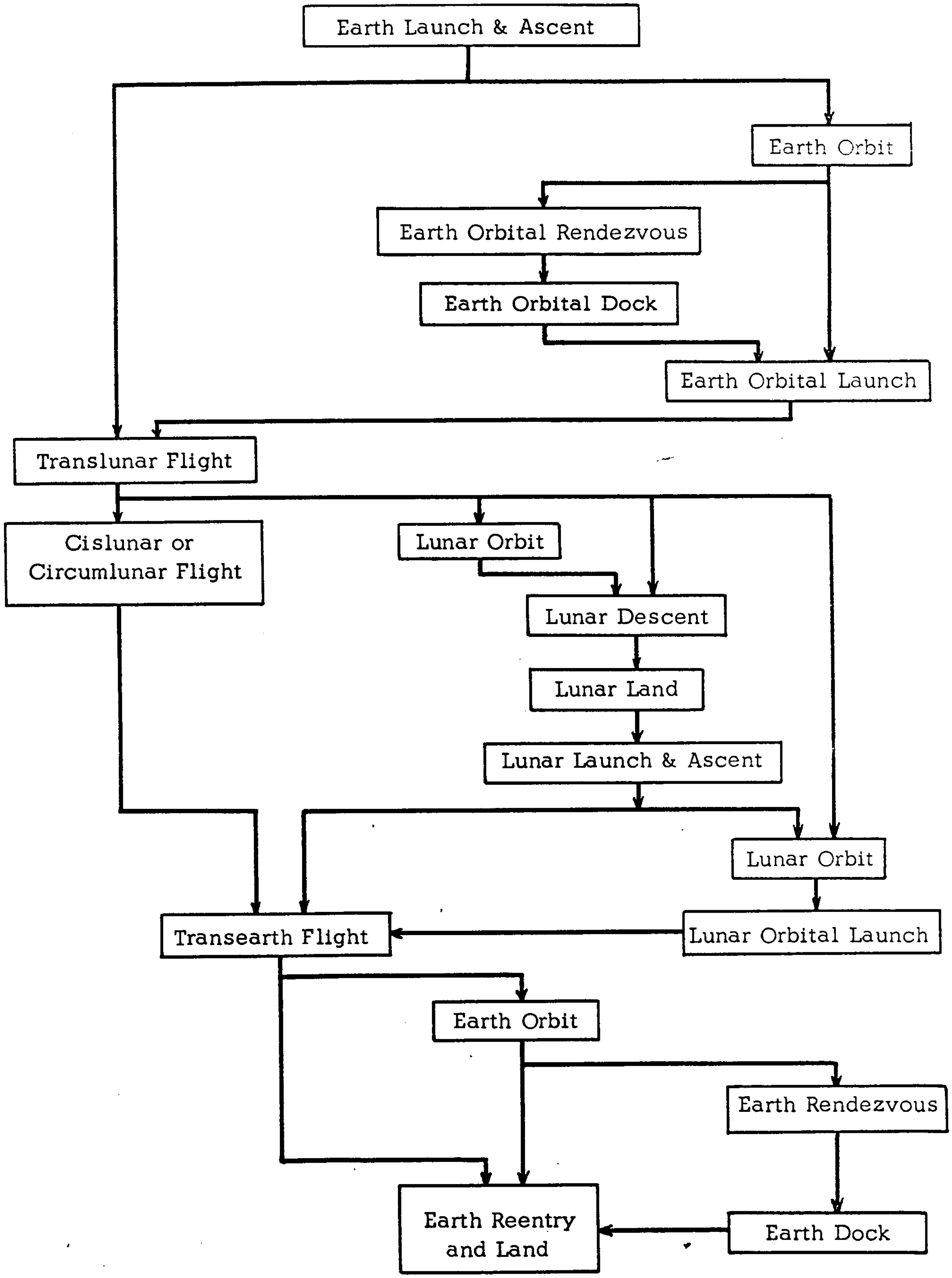

FIGURE 5. MANNED LUNAR MISSION FLIGHT SEQUENCES

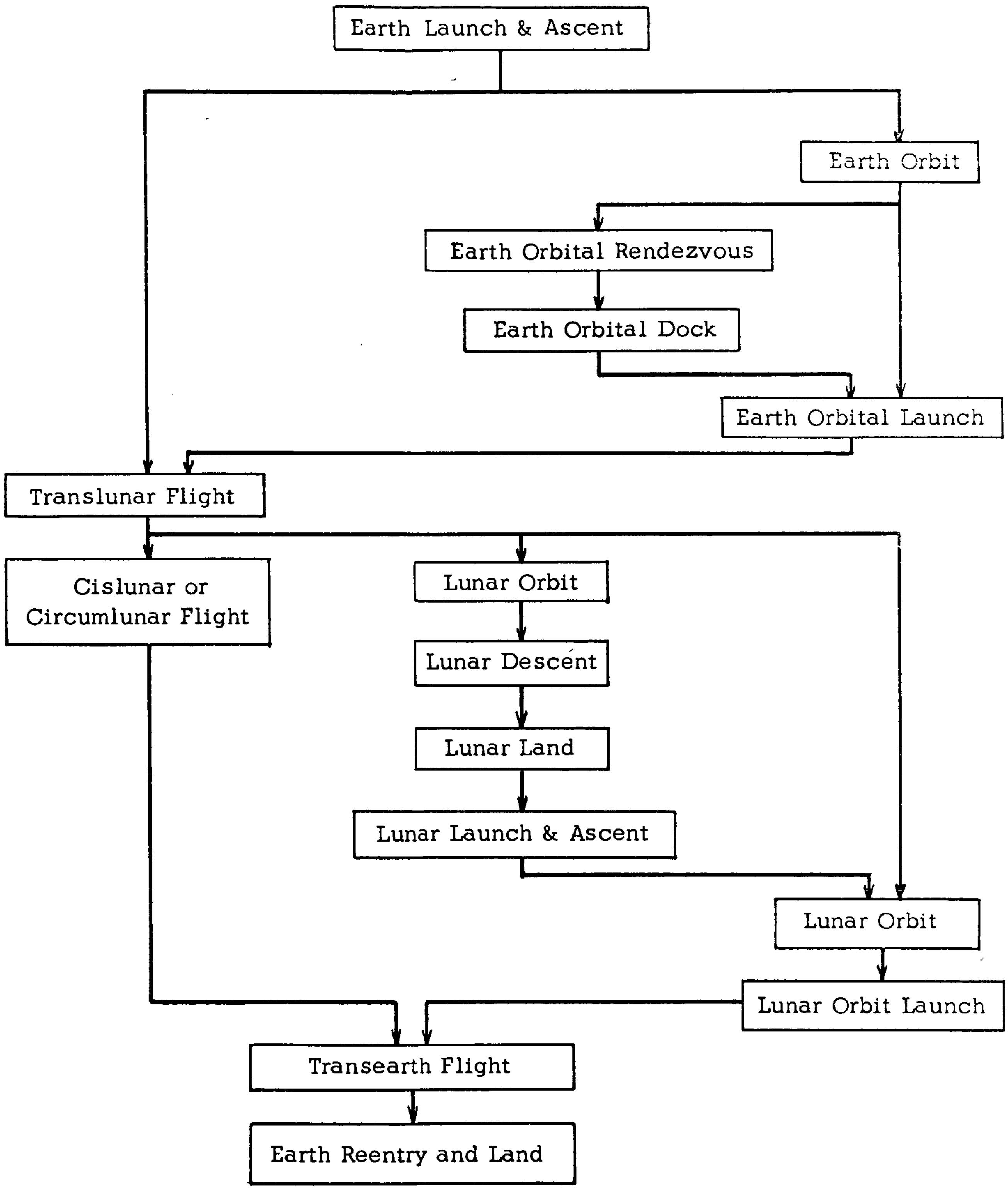

FIGURE 6. PROBABLE LUNAR MISSION FLIGHT SEQUENCES FOR FIRST GENERATION VEHICLES

EVENTS	SECONDARY MISSION
Launch Pad Checkout ↓	Earth Launch and Orbit
Earth Launch ↓	
Ascent ↓	
Parking Orbit Injection ↓	
Parking Orbit Sustenance ↓	
Pre-Orbital Launch Checkout ↓	Earth Orbital Launch and Translunar Flight
Earth Orbital Launch ↓	
Translunar Trajectory Injection ↓	
Translunar Trajectory Sustenance ↓	
Translunar Midcourse Correction ↓	
Translunar System Checkout ↓	
Lunar Orbit Injection ↓	Lunar Land
Lunar Orbit Sustenance ↓	
Lunar Descent ↓	
Lunar Hover and Land ↓	
Lunar Prelaunch Checkout ↓	Lunar Launch and Transearth Flight
Lunar Launch ↓	
Lunar Ascent ↓	
Lunar Orbit Injection ↓	
Lunar Orbit Sustenance ↓	
Lunar Orbital Launch ↓	
Transearth Trajectory Injection ↓	
Transearth Trajectory Sustenance ↓	
Transearth Midcourse Correction ↓	
Transearth System Checkout ↓	
Earth Orbit Injection ↓	Earth Re-entry and Landing
Earth Orbit Sustenance ↓	
Earth Re-entry ↓	
Earth Land	

TABLE 6. APOLLO MISSION AND EVENT SEQUENCE

guidance. Also, discrimination between manned and unmanned guidance is not made in this analysis although it is readily recognized and supported in numerous documents that manned supported guidance is more reliable and perhaps more accurate than unmanned.

GENERALIZED STATE-OF-THE-ARTS

4.10 Prior to discussing the guidance requirements associated with specific missions of the manned lunar program, an evaluation of generalized guidance system and subsystem state-of-the-arts seems appropriate. Information relative to the state-of-the-arts was obtained from various sources. Some values seem optimistic; although not indicated as such, the performances indicated probably represent that attained under near-ideal conditions and environment, or represent engineering judgement as to probable performances that can be obtained.

4.11 Table 7, presents an analysis of the current or near future guidance state-of-the-arts. This represents the capabilities that will be available for the early lunar missions. Some extension of these state-of-the-arts is anticipated based on current research and development programs, but until these extensions are proven, the values indicated in Table 7 should be considered representative of current capabilities.

MISSION GUIDANCE

Earth Launch and Orbit Mission:

4.12 Prior to launch there is a complete operational checkout of all major systems and subsystems to be utilized in the guidance of the launch vehicle during ascent and injection into the desired orbit, and maintaining this orbit. The philosophy of checkout will be to stimulate the systems in the same manner and sequence occurring in flight; this flight simulation checkout is included as part of the countdown procedures. Pre-launch checkout is a ground support function not to be considered in this study. However, checkout should establish the ability of the launch vehicle to maintain guidance during this mission to assure successful accomplishment of the mission.

4.13 The accuracy with which a vehicle can enter and maintain proper orbit depends on:

a. The accuracy of the impulse, a function of guidance and control.

[redacted]

[illegible]

AN ANAL[illegible] CE SYSTEM [illegible]
[illegible] THE-ARTS

[illegible]

[illegible]	[illegible] 1° - 0.25°
[illegible] Tracker	[illegible]
[illegible] Tracker	[illegible]°
[illegible] Gyros	0.1°/hr
[illegible]	[illegible] sec of arc/deg/[illegible]
[illegible] Platform	
[illegible] Free Fall Drift	[illegible]
[illegible]	[illegible]
[illegible]	[illegible]
[illegible]	
a. Bias	[illegible]
b. Scale Factor	[illegible]
[illegible] Altimeter (to [illegible])	1.0%
[illegible] Time	1 x 10[illegible]

Control Accuracies

1. Injection Angle Deviation	
a. Azimuth	0.02°
b. Pitch	0.02°
2. Velocity	5.0 fps

NOTE: These are 3σ accuracies unless otherwise noted.

[redacted]

b. The knowledge of instantaneous variables during impulse such as position, velocity, and time.

c. The knowledge and effect of perturbating forces, especially during the coast part of injection, and orbiting.

Requirements:

4.14 Ascent: Many of the systems utilized in ascent guidance will be activated in countdown to ensure system operation as soon as launch is initiated. During ascent, guidance requires the computation and monitoring of such parameters as velocity, position, and time. If the vehicle is to enter and maintain proper ascent trajectory, accelerometers will survey vehicle velocity and the inertial stable platform will monitor the orientation of the vehicle to within predetermined limits. These measured parameters are compared with programmed information by the guidance computer, which analyzes the information and forwards subsequent commands to the control system. Peturbations to the vehicles during ascent are monitored. Phenomena such as wind shear, engine out, control malfunction, fuel slosh, bending, vibration, aerodynamic buffeting and dynamic unbalance can cause such disturbances. The guidance and control sensors will analyze and command the control system to counter these inputs or abort the mission if they are too great. Characteristics and capabilities of the sensors and the disturbances are discussed in the Control Technical Area. These inputs to the guidance system sensors are generally filtered out so that guidance will result from flight parameters only. These disturbances are only monitored and analyzed by the guidance system.

4.15 The accuracy with which the ascent trajectory must be maintained is dependent on the accuracy with which the desired orbit is to be established, which in turn is dependent on the ultimate mission of the vehicle in orbit, and corrective capabilities of the vehicle in orbit or during later phases of the flight. Each "laxity" in system performance tolerated within a vehicle requires more stringent performance by another system in order to maintain over-all vehicle performance. Thus, in ascent, improved guidance will result in less stringent requirements on the corrective capabilities of the vehicle which is highly desirable for preservation of fuel, decrease in payload weight, etc.

4.16 In defining the guidance requirements during ascent, assumptions will be made as to the probable orbital missions and orbital characteristics desired. Ascent into a nominal 110 nautical mile circular parking orbit, transfer into a 300 nautical mile circular rendezvous or lunar launch orbit, direct ascent to a 300 nautical mile circular rendezvous or launch orbit, and direct injection into a translunar trajectory will be discussed.

Other considerations will be directed toward the orbit plane angles which may have to coincide with that of the moon or another orbital vehicle, and coordination of the time of launch with another event, such as rendezvous with another vehicle, ephemeris of the moon, etc.

4.17 A circular parking orbit, ranging in altitude from 100 to 125 nautical miles is preferred.1/2/3/4/ The orbit life at that altitude is compatible with the requirements of expected parking orbit missions. Also, a satellite at that altitude rotates around the earth 240° per hour; a satellite at a 300 nautical mile altitude rotates around the earth at 225° per hour. Thus, if vehicle rendezvous is to occur and the vehicles should be out of phase or have substantial angular displacement between them, this displacement can be decreased 15° per hour.

4.18 A typical ascent into orbit would have a first stage boost to a velocity of approximately 10,000 fts and a second stage boost to a nominal orbital velocity of 25,500 to 26,000 fps.

4.19 The first stage boost would last 150-160 seconds, and end at an altitude of 35 nautical miles and a slant range of 50 nautical miles. The flight angle would be 23-25°. The dynamic pressure could approach 12 pounds per square inch; this phase of ascent encompasses the high dynamic pressure region. This stage is burned till the propellant is depleted, then separation occurs.

4.20 Accuracy requirements during first stage boost are not too severe. Corrections can be made during the second stage boost at a moderate cost in performance. During the first stage, the ascent trajectory is compared to the reference trajectory—errors or deviations are noted. If they are too great, it may be an indication that abort may be necessary.

1/ North American Aviation, Inc., Space and Information Systems Division, Final Report NASA Study of Large Launch Vehicle Subsystems, Oct. 1961, CONFIDENTIAL.

2/ Lockheed Georgia Co., Lockheed Aircraft Corp., Study Report Criticality of Subsystems for Large Launch Vehicles, 7 Oct. 1961, CONFIDENTIAL.

3/ Marshall Space Flight Center, NASA, Huntsville, Ala., Orbital Operations Preliminary Project Development Plan, 15 Sept. 1961, CONFIDENTIAL.

4/ Missile and Space Division, Lockheed Aircraft Corp., Sunnyvale, Calif. Final Report Orbital Docking Test Study, 26 June 1961, CONFIDENTIAL.

4.21 The second stage thrust to orbital velocity for parking orbit would be approximately 7 minutes of continuous boost to injection.

4.22 Requirements for entering a parking orbit with extreme accuracy in altitude seems quite unnecessary; thus a 110 $\pm$ 10 nautical mile limit on parking orbits is adequate. This capability is well within the capabilities of current guidance techniques. Figures 7, 8, and 9 present the anticipated history of the Apollo Launch Vehicle during the ascent to 100 nautical mile circular parking orbit.

4.23 Although the altitude requirements for a parking orbit may not be rigid, the orbital plane angle requirements, again depending on the vehicle mission, may be quite severe. For rendezvous with another vehicle, or launch to the moon from the parking orbit, matching of the orbit plane with a desired plane may be necessary to ensure mission success. This can be done by injecting the vehicle into the proper plane on launch, or by orbital plane transfer (doglegging). For propulsion economy, doglegging is limited to approximately 5° (or $\Delta v \sim 500$ fps) angular displacement. Since it requires substantial propulsive thrust to perform doglegging, it becomes less desirable than accurate injection into the desired plane. If at all possible, the necessary orbit plane should be attained to within 1.0° ($\Delta v \sim 100$ fps) for rendezvous. The penalty, or additional velocity increment needed for transfer into a new plane has been analyzed for a vehicle at an altitude of 263 nautical miles;[5] Figure 10 relates the velocity increment required for plane changes.

4.24 Direct injection into the circular 300 nautical mile lunar launch or rendezvous orbit will result in more stringent requirements of guidance especially if rendezvous is to occur with a vehicle already in orbit. Vehicle velocity will have to be maintained to within 5 feet per second, and plane angle deviation to 0.5°. To do this, altitude will have to be determined to 1×10^{-3}, acceleration to 1×10^{-4}, time to 1×10^{-6}, attitude reference to 0.1°, and azimuth to 0.1°.[6]

4.25 The accuracy requirements associated with direct injection from earth into translunar trajectory are of the same order of magnitude associated with orbital injection into translunar flight which are discussed as part of the translunar mission. With launch from earth rather than orbit, launch time and direction becomes much more critical.

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[5] NORAIR Division, Northrop Corp., Hawthorne, California, Preliminary Parametric Analysis for an Orbital Rendezvous Base System, 19 October 1961, UNCLASSIFIED.

[6] Lockheed Georgia Co., op. cit.

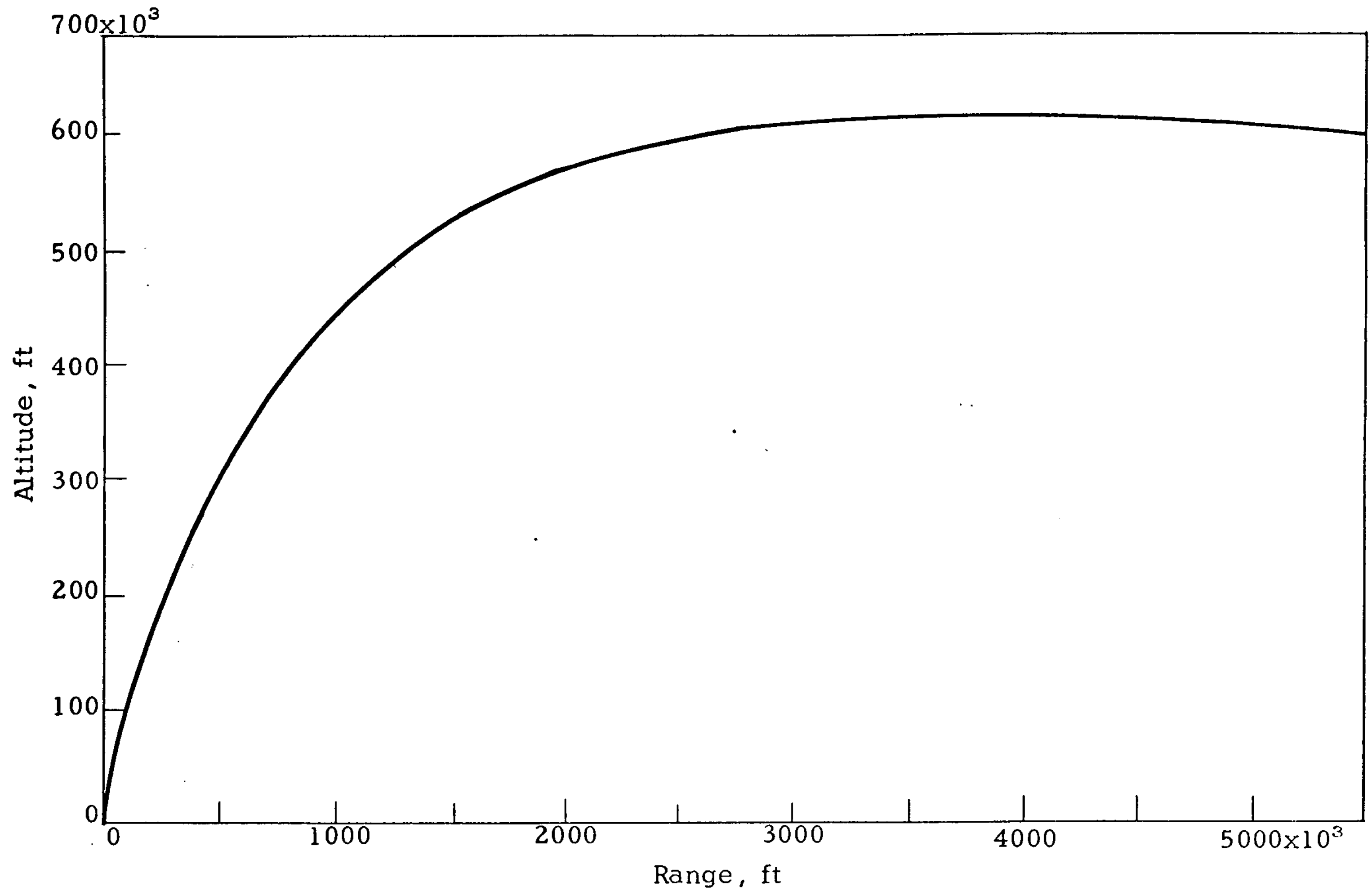

FIGURE 7. ALTITUDE VS. RANGE FROM LIFT-OFF TO PARKING ORBIT (APOLLO)
Launch Azimuth = 90° (Due East)

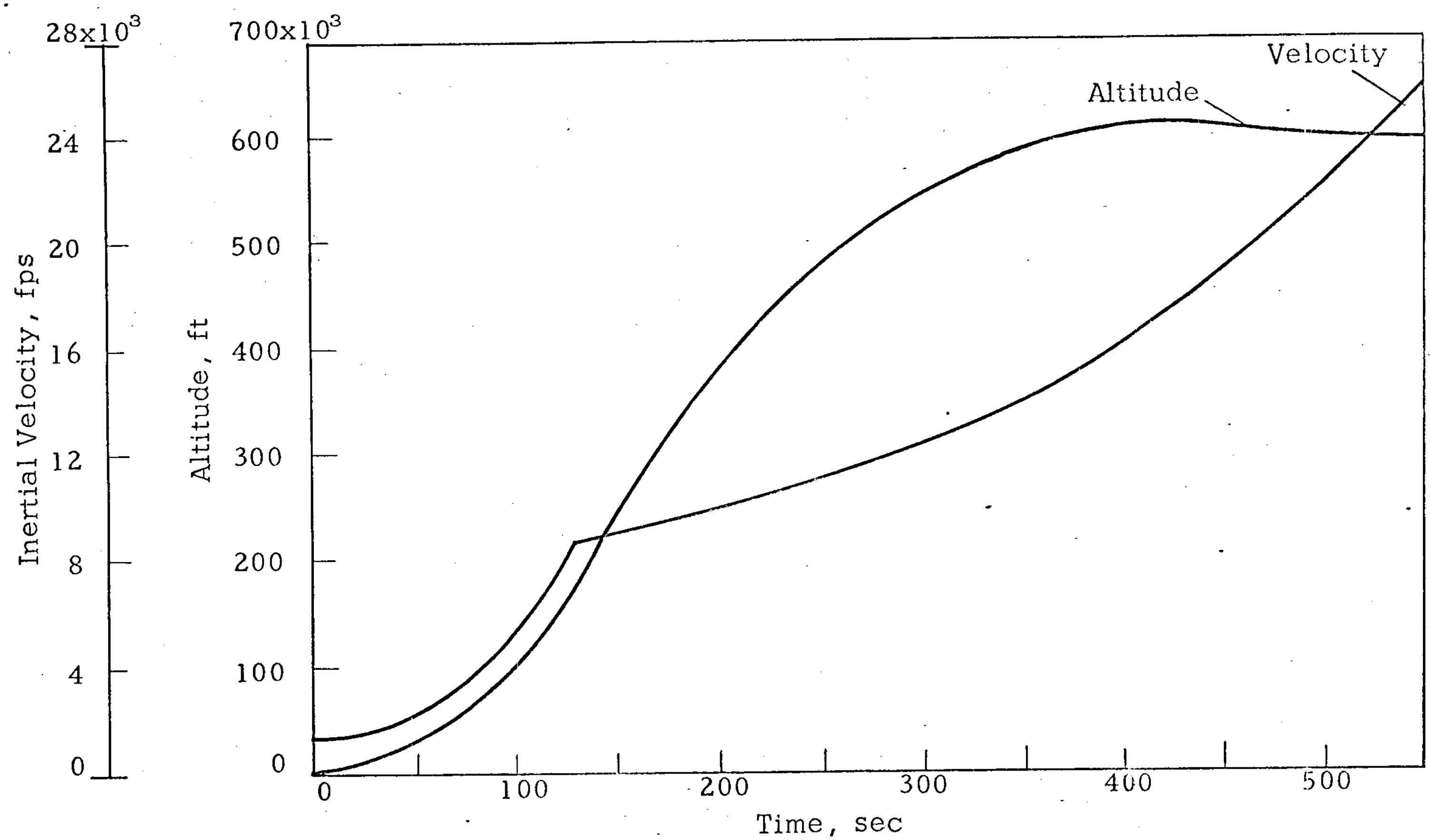

FIGURE 8. TIME HISTORY FROM LIFT-OFF TO PARKING ORBIT FOR APOLLO

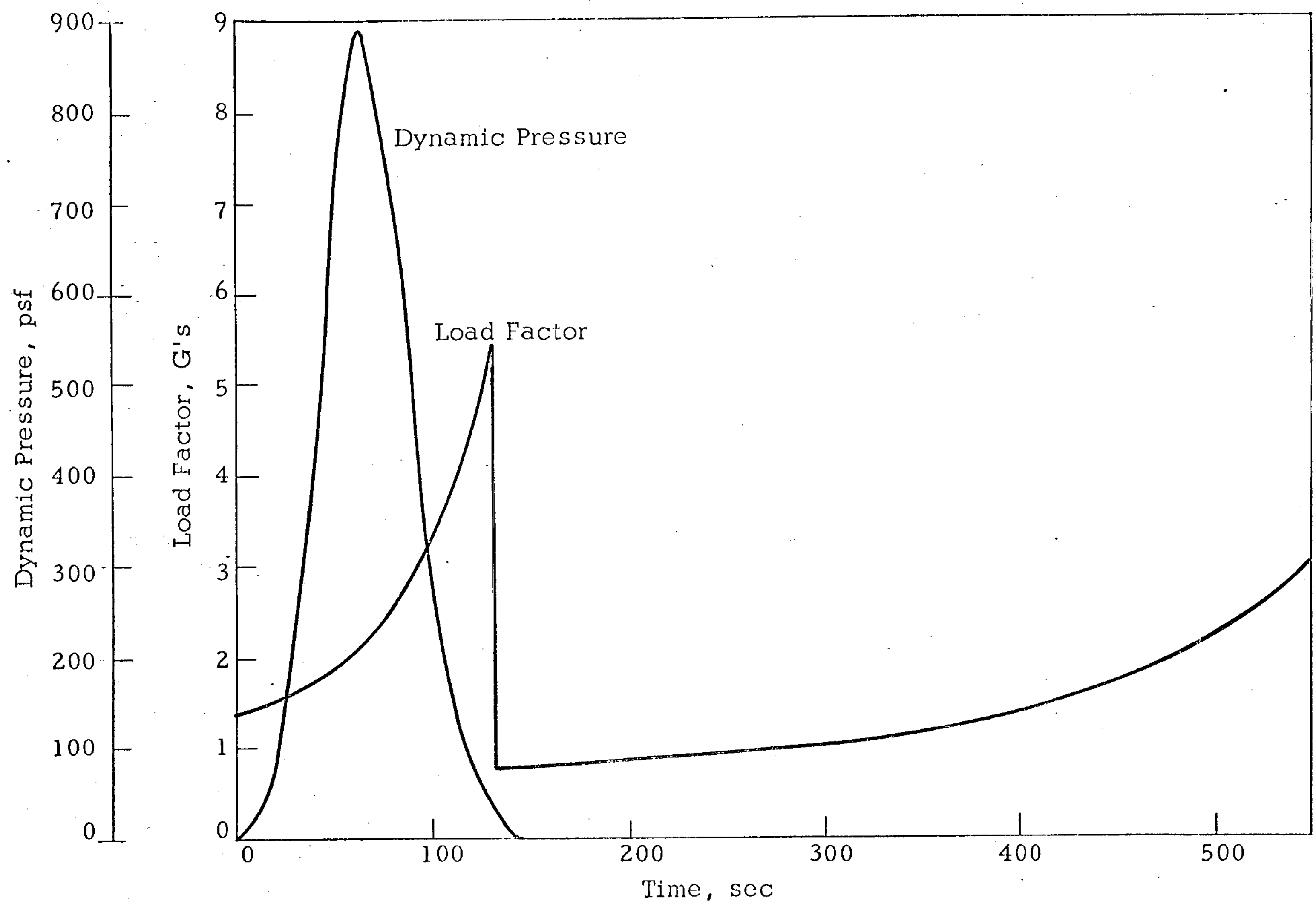

FIGURE 9. TIME HISTORY FROM LIFT-OFF TO PARKING ORBIT FOR APOLLO

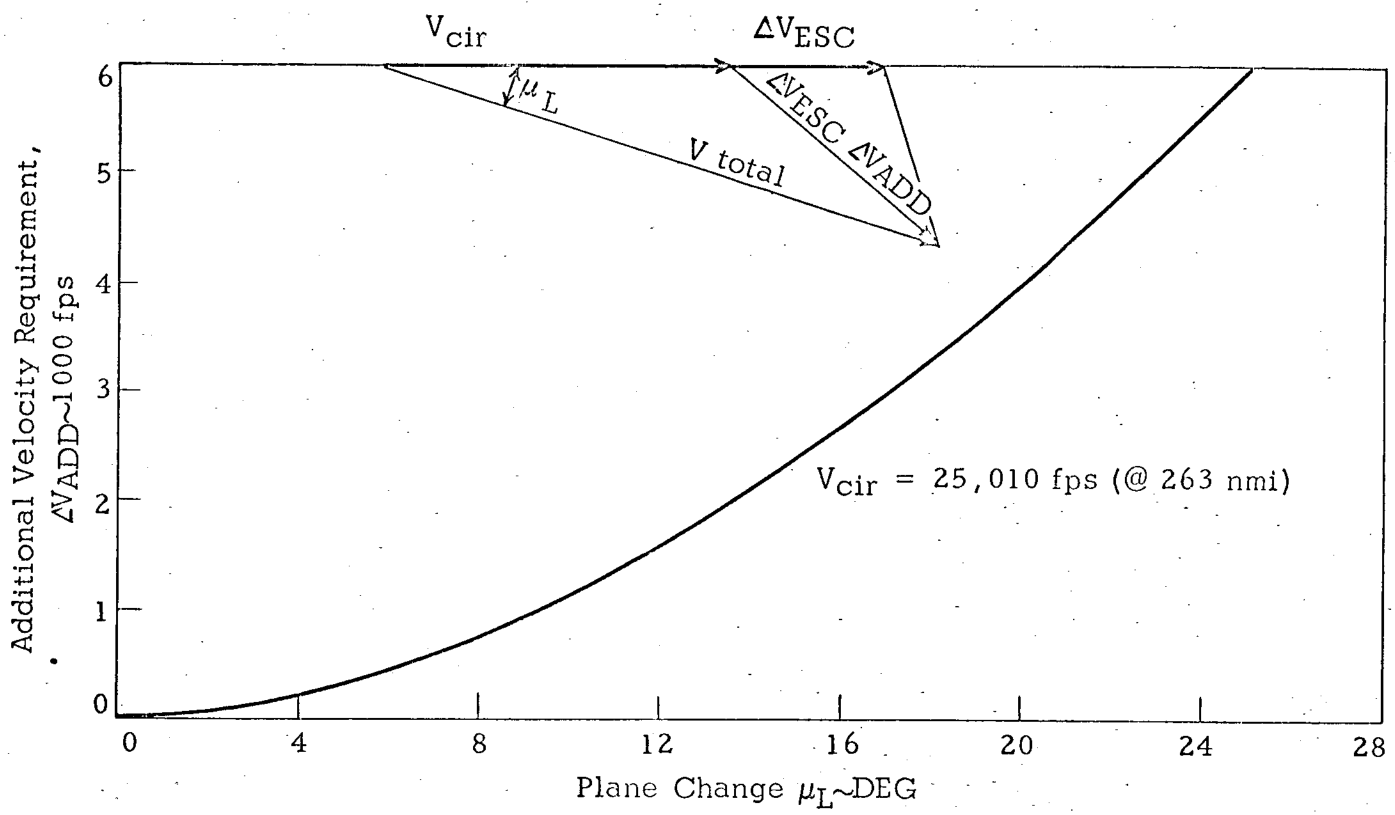

FIGURE 10. VELOCITY INCREMENT INCREASE DUE TO P'ANE CHANGE REQUIREMENT

4.26 A three dimensional study[7] analyzed the accuracy requirements of lunar trajectories to strike (hard-land) the moon, for three dates: November 7, 15 and 23, 1959. The analysis assumed a vehicle burnout altitude of 388.5 statue miles and a burnout velocity 7.177 statue miles per second. Only one error at a time was assumed present. The results are listed as:

Parameter	Allowable Spread Nov. 7	Nov. 15	Nov. 23
Altitude (st. miles)	(-14.5 to 21.2)	(-.18 to 22.0)	(-1.9 to 55)
Velocity (fps)	(-79.0 to 101.0)	(-10.0 to 82.0)	(-9.9 to 124)
Earth Angular Displacement (degrees)	(-.202 to .075)	(-.43 to .06)	(-.067 to 1.56)
Aximuth (degrees)	(-.202 to .075)	(-.03 to .32)	(-.28 to .41)
Flight Path (degrees)	(Not given)	(-.075 to .264)	(-.35 to .002)

This hard land on the moon will occur at any point on the moon; the point of impact was not controlled in this analysis. This data shows the significance of the earth/moon position in the trajectory requirements of the vehicles, indicates the magnitude of the allowable spread in parameters, and reveals the presence of significantly fluctuating requirements.

4.27 The probability for necessary abort is greatest during the launch or ascent phase, and guidance will have to be decided when and if to abort. Conditions justifying abort are:

a. Too great a trajectory deviation.

b. Boost stage does not ignite.

c. Sensors indicate possiblity of explosion.

d. A critical system becomes inoperative.

e. The autopilot becomes unstable.

4.28 Orbital Transfer: Vehicle transfer from circular orbit at one altitude to a circular orbit at a second altitude, expected to be Hohmann (minimum energy ellipse) transfer, is another ascent (or descent) and orbit injection consideration. For transfer, the ephemeris of the present orbit must be determined, the characteristics of the new orbit must be established,

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7/ A. Petty, I. Jurkevich, M. Fabrize, and T. Coffin, Lunar Trajectory Studies, General Electric Company, Missile and Space Vehicle Dept., AF CRL 507, dated June 1961, UNCLASSIFIED.

then the guidance system will then calculate the velocity boost (or retro for descent)/time relationship necessary for the orbital transfer. Again, accuracies are a function of mission. If rendezvous is to occur following transfer, then the accuracies will be expected to be comparatively severe.

4.29 Orbital transfer techniques will utilize the digital computer, stable platform, signal processor, control computer, and for early missions, the command link. Data to be established prior to transfer are: angular data for altitude, magnitude of the velocity increment, and the time for ignition. The magnitude of the impulse must be held to very close tolerances; the direction of impulse is insensitive to small errors; thus, is not so critical.

4.30 During transfer for rendezvous, the position of the target vehicle with respect to the chaser will have to be established and monitored constantly; the transfer will be coordinated with this relative positioning. Acquisition of the target vehicle will probably occur at a range of 400-600 nautical miles, the expected radar range.[8/][9/]

4.31 To transfer from a circular 110 nautical mile parking orbit to a 300 nautical mile rendezvous or launch orbit involves a boost of 400-500 feet per second at the beginning of transfer, a midcourse correction capability of 150 feet per second, and a boost of 400 to 500 feet per second to circularize the orbit at 300 nautical miles. The operation will take approximately 45 minutes to 1 hour, in time and 180° of angular displacement around the earth. It will begin when the vehicles have 7° or less of earth angular displacement between them; the lower altitude vehicle lagging the higher altitude vehicle up to 7.1° or leading by no more than 6.4°.[10/][11/]

4.32 Through transfer, the line of sight must be established to $\pm$ .01°, range between vehicles to .1%, a range rate to 1 foot per second, and velocity to $\pm$ 5 feet per second. The attitude, acceleration, time, altitude, and injection azimuth errors should be the same as that discussed previously for direct injection into a 300 nautical mile orbit.[12/]

8/ Lockheed Georgia Co., _op. cit._

9/ General Dynamics/Astronautics, _A Study of Large Launch Vehicle Systems for a Manned Lunar Landing Program_, 9 October 1961, CONFIDENTIAL.

10/ Lockheed Georgia Co., _op. cit._

11/ H. A. Lieske, Rand Corporation, "Accuracy Requirements for Trajectories in the Earth-Moon System," _Vistas in Astronautics_, Vol. 1, 1958, UNCLASSIFIED.

12/ Lockheed Georgia Co., _op. cit._

4.33 The accuracies to which the ephemeris must be established prior to transfer have been analyzed by NASA.[13] Assuming an acceptable error of 8 km in position following transfer, these accuracies are:

Velocity	0.9 m/sec or 3 ft/sec
Path Angle	0.01 degrees
Azimuth Angle	0.13 degrees
Altitude	0.9 km or 0.5 nm
Position along Orbit	11.4 km or 6 nm
Position along Perpendicular to Orbit	3.3 km or 2 nm

4.34 If the transfer is to be coordinated with the flight of another vehicle, premature or late initiation of the transfer could have disasterous results on the mission. Premature transfer could necessitate substantial retro or slow down of the vehicle during transfer or after it was in orbit; late transfer could necessitate substantial boost to catch up to the other vehicle, only to be followed by retro to slow down to rendezvous velocity.

4.35 Maintaining Orbit: The time which a vehicle remains in orbit is dependent on the vehicle mission and the vehicle orbit altitude of velocity. In a parking orbit at an altitude of 100-125 nautical miles, the vehicle velocity is approximately 25,500 feet per second and the life of the vehicle is a minimum of 40 hours without corrective boost. In a 300 nautical mile circular orbit the vehicle has a minimum normal life span of two years at a velocity of 26,000 feet per second. The approximate rate of altitude decrease is 5 to 50 km and .001 to .050 km per day at the lower and higher altitudes, respectively. The vehicle slows down due to atmospheric drag, a function of the vehicle drag coefficient, atmospheric density, vehicle cross-sectional area and the vehicle mass. As it slows down, it drops to the earth into more dense atmosphere, which slows it down additionally, etc. Perturbations to the earth orbits are due to many

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[13] NASA, Huntsville, Ala., *op. cit.*

phenomena: earth oblateness,14/ lunar and solar effects,15/ 16/ gravitational anomalies, and from guidance injection errors. Depending on the subsequent mission of the vehicle or the accuracy with which it was attained, it may be necessary to change the plane of the orbit (dogleg) during the vehicle orbital flight.

4.36 If the vehicle begins to descend to earth, the guidance and tracking sensors will have to detect this change in orbit in order to supply corrective boost. Also, since this mission can be a long term mission, errors due to drift of the inertial guidance system become significant and realignment of the stable platform is required. Regression of the orbit plan occurs due to oblateness of the earth and the other perturbations mentioned previously; this occurs in a direction opposite to rotation of the satellite around the earth and is a function of the inclination of the satellite plane to the equator as shown in Figure 11. For inclination of 30° and a 110 nautical mile orbit, the regression rate would be $.48^{\circ}$/satellite revolution; for a 300 nautical mile orbit, the regression rate would be $.44^{\circ}$/satellite revolution. In maintaining orbit around the earth, it is very important that the ephemeris of the orbiting vehicle be established and constantly monitored accurately. For orbital missions such as orbit transfer, lunar launch, orbital rendezvous, and orbital docking, the orbit parameters should be established to the following limits: 17/

Vehicle Velocity	± 3 feet/sec.
Altitude	± 0.5 nautical miles
Position along orbit	± 5 nautical miles
Azimuth Angle	± .01 degree
Path Angle	± .01 degree
Normal distance from orbit plane	± 2 nautical miles

14/ Earth oblateness caused inaccuracies of up to 0.1° in establishing the attitude of earth oriented bodies.
U.M. Hatcher and E.F. Germain Jr, Study of a Proposed Infrared Horizon Scanner for Use in Space Oriented Control Systems, NASA, TN-D-1005, Jan. 1962.

15/ G.E. Cooke, Luni-Solar Perturbations of the Orbit of an Earth Satellite, Royal Aircraft Establishment, Technical Note No. G.W. 582, July 1961.

16/ The Expected Life of Explorer VI was shortened by a factor of ten by Solar and Lunar Pertubations.
Peter Musen, On the Long Period Luni-Solar Effect in the Motion of an Artificial Satellite, NASA, TN-D-1041, July 1961.

17/ NASA, Huntsville, Ala. op. cit.

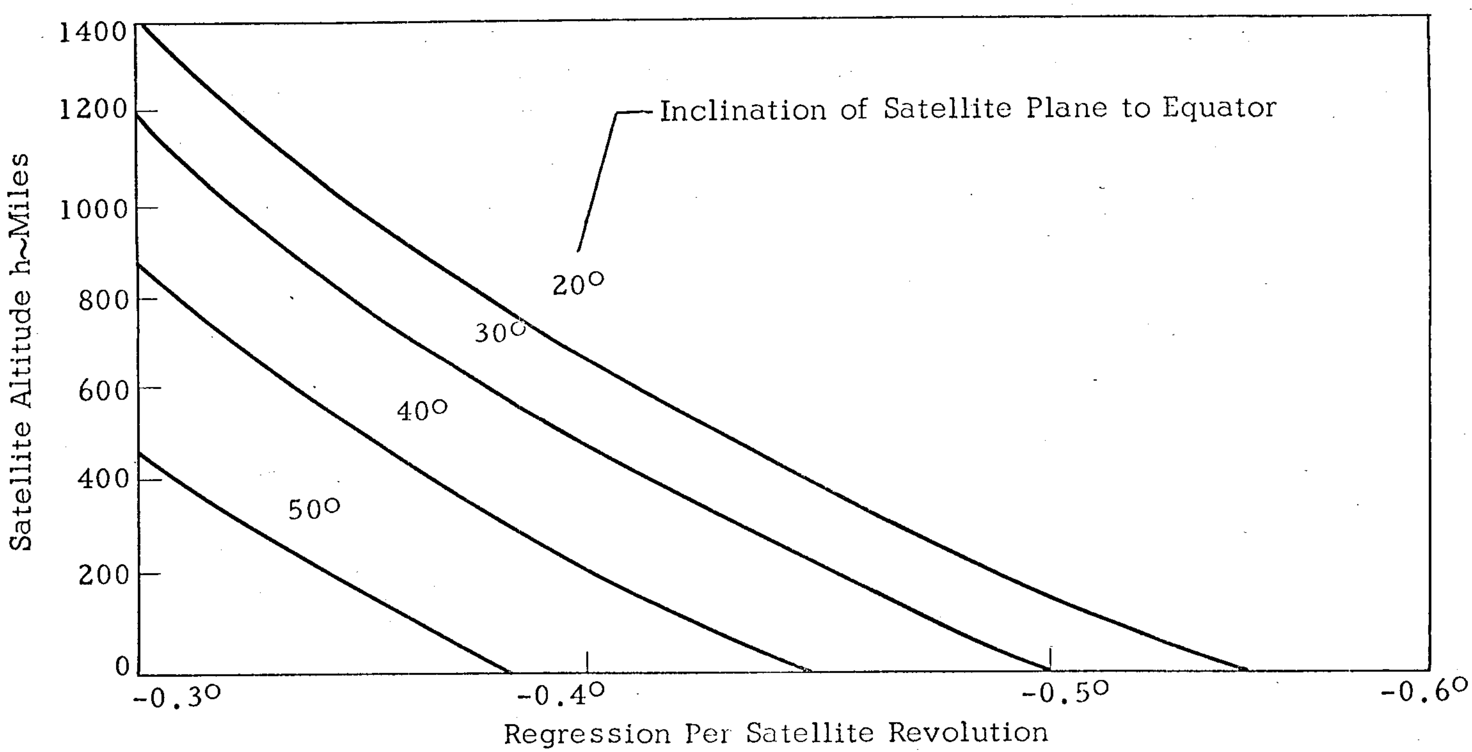

FIGURE 11. REGRESSION RATE OF AN EARTH SATELLITE IN A CIRCULAR ORBIT

State-of-the Arts

4.37 As mentioned previously, guidance establishes vehicle position, velocity and time; it involves not only the sensing and measurement of these parameters, but the subsequent analysis and control of them. In discussing the guidance state-of-the-arts all of these functions will be be considered and reflected in the evaluation.

4.38 Radio guidance systems, similar to that in use at the Atlantic Missile Range (AMR), Cape Canaveral, Florida exceed all guidance requirements for this mission; however, this technique is hampered by line-of-sight considerations. Substantial ground support would be required throughout the world; thus, inertial systems backed up with a tracking system will probably be used.

4.39 It is within the present technical capabilities to maintain an injection velocity of approximately 26,000 feet per second to $\pm$ 5.0 feet per second using precision accelerometers although $\pm$ 2.5 feet per second may be attainable in the very near future. Injection or ascent angles can be held to $\pm .02^{\circ}$ using inertial systems.[18] This is adequate for orbital operations expected within the near future. It is anticipated that improvement in inertial system and accelerometer performance will provide the improvement for lunar operations within the next few years.

4.40 Typical precision accelerometers are accurate to 1.0×10^{-4}g or better. Guidance system drift during injection would range from .002 to .02 degrees based on estimated acceleration sensitive drift of 0.1 to 1.0 $\times 10^{-3}$ degrees per thousand feet per second of velocity added by thrusting, mass unbalance drift coefficients to .01 to 0.1 degrees per hour per g., and anisoelastic drift of .02 degrees per hour per g^2.[19]

4.41 The guidance accuracy requirements for injection are within the state-of-the-arts, and very nearly satisfied by systems being delivered, such as the Centaur, Atlas, Saturn, and the Minuteman inertial systems.[20]

4.42 A summarization of guidance accuracy requirements for the Earth Launch and Orbit Mission is presented in Table 7a. The requirements indicated are within the state-of-the-arts.

Orbital Rendezvous Mission

4.43 Orbital rendezvous begins with the acquisition of the target vehicle by the chaser vehicle, and ends with completion of terminal guidance or the beginning of the docking phase, at which time the vehicles are coasting

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[18] Lockheed Georgia Co., op. cit.

[19] Lockheed Georgia Co., op. cit.

[20] North American Aviation, Inc., op. cit.

TABLE 7a

A SUMMARIZATION OF GUIDANCE REQUIREMENTS
FOR THE EARTH LAUNCH AND ORBIT MISSION

EVENT	PARAMETERS	ACCURACY REQUIREMENTS
Parking Orbit Ascent	1. Velocity	—
	2. Altitude (control)	± 10 nm
	3. Angular Deviation (Azimuth and Pitch)(control)	± 1°
	4. Time	—
Rendezvous or Launch Orbit Ascent	1. Velocity (control)	± 5 fps
	2. Altitude (measurement)	± 0.1%
	3. Azimuth (control)	± 0.1°
	4. Pitch (control)	± 0.1°
	5. Plane Angle Deviation (control)	± 0.5°
	6. Attitude Reference (measurement)	± 0.1°
	7. Time (measurement)	± 0.0001%
	8. Acceleration (measurement)	± 0.01%
Orbit Transfer	1. Velocity (control)	± 5 fps
	2. Altitude (measurement)	± 0.1%
	3. Azimuth (control)	± 0.1°
	4. Attitude Reference (measurement)	± 0.1°
	5. Time (measurement)	± 0.001%
	6. Acceleration (measurement)	± 0.01%
	7. Target Acquisition Range (measurement)	0.1%
	8. Target Range Rate (measurement)	1 fps
Maintaining Orbit	1. Velocity (control)	± 3 fps
	2. Altitude (measurement)	± 0.5 nm
	3. Azimuth (measurement)	± 0.01°
	4. Path Angle (measurement)	± 0.01°
	5. Position along Orbit (measurement)	± 5 nm
	6. Normal Distance from Orbit Plane (measurement)	± 2 nm

together. Both vehicles could be unmanned—under such circumstances rendezvous would probably be controlled by ground support commands, or possibly by command link to another manned platform or vehicle. At times the chaser may be manned, or the target vehicle—or both vehicles. If the chaser vehicle acquires the target vehicle from a parking orbit, Hohmann or minimum energy orbit transfer, could be included as part of this mission. The requirements associated with orbit transfer were discussed in the previous section. Guidance during this mission is divided into two separate phases: coarse and terminal. The coarse, or initial phase, places the chaser vehicle at a distance at which terminal guidance techniques can begin. Essentially the same launch vehicle systems are utilized in both phases of guidance; however, the capabilities of the systems for terminal guidance are generally more refined.

4.44 In rendezvous guidance, the tracking radar system provides data to the guidance system to compute range, range rate, line-of-sight angular displacement, and line-of-sight angular displacement rate. The computer uses the radar antenna angles as information for generating commands for attitude control maneuvering. Accelerometer outputs are used primarily as line-of-sight angular rate and thrust termination determinators. The data are analog (although digital accelerometers are being developed) and are digitized for computation and analysis (compared with programmed trajectories). Digitized analyses are converted to analog form and are forwarded as instructions to the control system.[21]

4.45 The computer is essentially the brain of the chaser, it provides solutions to the terminal guidance equations and handles much of the mission sequencing. The closing rates are computed and monitored by the computer; braking is accomplished through the thrust reversal or retro system. On command from the computer, the control system nulls line-of-sight angular rates and maintains this null orientation. The horizon sensor establishes a common roll reference between the two vehicles. The optical alignment system provides accurate attitude alignment along the line-of-sight during the final phase of terminal guidance. A television camera would assist in manned rendezvous.

Requirements:

4.46 <u>Coarse Guidance</u>: The requirements of coarse guidance need not be specified to close tolerances; there is adequate time to adjust and correct to close tolerances during terminal guidance.

4.47 Coarse guidance should acquire the target over a short range of at least 400 to 600 nautical miles, and guide the chaser to within a slant

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21/ W.E. Brunk and R.J. Flaherty, <u>Methods and Velocity Requirements for the Rendezvous of Satellites in Circumplanetary Orbits</u>, NASA, TND-81, Oct. 59.

range of 5 to 10 nautical miles of the target. This phase could also include the orbit transfer discussed in the previous section. Assuming proper functioning, this phase should last 45 to 70 minutes. The position of the chaser vehicle with respect to the target vehicle at the completion of this phase is somewhat arbitrary; it should be compatible with the capabilities of terminal guidance of the vehicle.

4.48 However, it is extremely important to establish the flight parameters of both vehicles accurately in order that terminal guidance can be carried out accurately to ensure maximum fuel economy. Flight path errors of 0.1° to 1.0° require a velocity change of approximately 100 fps; velocity errors of up to 10 fps are compensated by velocity change of approximately 70 fps. Required accuracy of measurement will be discussed later.

4.49 Terminal Guidance: Terminal guidance systems should operate omnidirectionally over a slant range of 5 to 10 nautical miles. The closing rate of the vehicles should be such as to complete this phase in no more than 15 minutes. The difference in vehicle velocities within 1000 feet of rendezvous should not exceed 10 ft/sec. This closing velocity should be established to within 1 ft/sec. Optical alignment and guidance should also be used for the last 100 feet of separation till the docking phase begins. The last corrections to altitude, velocity, and position should be made by this time. Optical alignment errors should not exceed 1.0° in roll, and 0.1° in pitch and azimuth. Axial displacement should not exceed 1 foot at the end of terminal guidance.

Range Rate Accuracy:	1% or ± 1 ft/sec
Angle of Vision:	$\pm 90^{\circ}$.

4.50 SATURN radar altimeter is good to 450 km with velocity (altitude rate) and altitude errors of 8 meters per second and 30 meters, respectively.[22/][23/]

4.51 Parameters of rendezvous vehicles should be measured to the following accuracies.[24/]

Velocity	.1 meter per second
Path angle	.001 degrees (terminal)

22/ North American Aviation, Inc., op. cit.

23/ NASA, Huntsville, Ala., op. cit.

24/ NASA, Huntsville, Ala., op. cit.

Azimuth angle	.001 degrees (terminal)
Altitude	.10 km
Positon along orbit	10 km
Normal position to orbit plane	10 km

State-of-the-Arts:

4.52 Infra-red horizon sensors will be used for attitude reference; it will establish the horizon to within 0.1°. Thus, total reference misalignment could be 0.2°.

4.53 Doppler continuous wave tracking techniques proposed for use in vehicle rendezvous supposedly establish over a range of 1000 km, relative vehicle velocities to 1.0 meter per second, path angle to 0.1°, azimuth angle to $.01^{\circ}$, altitude to 1.0 km, position along orbit to 10 km, and position out of nominal orbital plane to 10 km.

4.54 Currently used pulsed radar techniques have, in general, significantly greater errors. For ranges of 100 (X-Band Radar) and 600 (Km Band Radar) nautical miles, range can be determined to $\pm$ .05%, range rate to $\pm$ 1 fps, angle to $\pm 0.1^{\circ}$ and angle rate to $\pm 0.02^{\circ}$/sec.[25]

4.55 Terminal Guidance Radar (X-Band) with a range of 50 ft to 6 nm, can establish range to $\pm$ 1 ft, range rate to 0.1 fps, angle to $\pm 0.06^{\circ}$ and angle rate to $\pm 0.006^{\circ}$/sec. C and L Band Systems with a range of $<$ 1 ft. to 4 n. miles, can establish range to $\pm$ 3 ft, range rate to $\pm$ 0.1 fps, angle to $\pm 0.06^{\circ}$ and lateral displacement at a range of 1 ft to 0.1 ft.

4.56 For ranges of 100 ft. to 2000 miles, the following reflects the radar altimeter state-of-the-art for measuring various parameters:[26]

Range Accuracy:	1% or $\pm$ 5 ft.
Range Rate:	1 ft/sec to 600 ft/sec

Table 8 summarizes the rendezvous tracking accuracy capabilities and requirements.

Orbital Docking:

4.57 Docking, which begins when terminal guidance of rendezvous ceases, covers the time period of approach or coasting together, and

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[25] General Dynamics/Astronautics, op. cit.

[26] General Dynamics/Astronautics, op. cit.

[redacted]

TABLE 8

ANALYSIS OF RENDEZVOUS TRACKING ACCURACY
CAPABILITIES AND REQUIREMENTS

	Doppler CW Tracking (100 km)	Pulsed Radar (100 nm X-Band)	Pulsed Radar (600 nm Km-Band)	Pulsed Radar (50 ft-6 nm X-Band)	Pulsed Radar (<1 ft to 4 nm C & L Band)	Requirements
1. Range	—	$\pm$ .05%	$\pm$ .05%	$\pm$ 1 ft	$\pm$ 3 ft	0.1%
2. Range Rate	1.0 m/s	1 fps	1 fps	0.1 fps	$\pm$0.1 fps	0.1 m/s
3. Angles						
a. Azimuth	0.01°	$\pm 0.1^\circ$	$\pm 0.1^\circ$	$\pm 0.06^\circ$	$\pm 0.06^\circ$	0.001°
b. Path	0.1°	$\pm 0.1^\circ$	$\pm 0.1^\circ$	$\pm 0.06^\circ$	$\pm 0.06^\circ$	0.001°
4. Angle Rates	—	$\pm 0.02^\circ$/sec	$\pm 0.02^\circ$/sec	$\pm 0.006^\circ$/sec	—	—
5. Attitude	1.0 km					0.1 km
6. Orbital Position	10 km					10 km
7. Normal to Orbit	10 km					10 km

[redacted]

subsequent mating of the vehicles. The functions of guidance in this mission are: to monitor the flight conditions initiated in rendezvous terminal guidance, to measure and monitor range and range rate between vehicles, ascertain the need for aborting the mission should conditions exist that may require such a decision, and maintain guidance of the mated assembly.

4.58 Definitions of the scope of orbital docking seems to vary; some definitions include terminal guidance as part of this mission, which for the study is considered a rendezvous function.

Requirements:

4.59 It is difficult to establish definite parameters under which orbital docking will occur. The primary requisite of the mission is to bring together two vehicles in orbit resulting in no damage to either vehicle or payload. To pinpoint the altitude at which it occurs, the distance between vehicles when it begins, the closing rate, and the accuracies in measuring these values, seems irrelevant to the mission just as long as the vehicles are engineered to accept the maximum parameters expected. However, an attempt will be made to discuss the most probable ranges of conditions under which docking will occur in earth orbit, earthbound or lunar bound.

4.60 The vehicles, following rendezvous terminal guidance, will essentially be in the same nominal orbit, approximately 5 to 25 meters apart; the closing rate should not exceed 3.0 to 5.0 meters per second; the closing period should last no more than 30 seconds. Lateral displacement at mating should not exceed 0.2 meters. Angular alignment of the vehicles within 1° will probably be necessary.

4.61 Because of the short ranges involved during this mission, just prior to contact of vehicles, pulsed radar techniques will probably not be able to provide necessary information, such as range and range rate, with the accuracy and information rate desired. Television, optical, or short range micro-wave techniques involving triangulation could be considered. The closing rate should be established to within 0.3 meter (1 foot) of the true values down to .3 meters (1 foot) from contact. Alignment of the vehicles will probably have to be monitored using optical techniques. Optical alignment techniques, with expected errors in roll, pitch, and azimuth of 1.0°, 0.1°, and 0.1°, respectively, will be required.

4.62 An analysis[27] of docking errors using persimistic inputs to the guidance systems such as three sigma range errors of 9 meters, velocity errors of 0.5 meter/second, and roll misalignment of 5 degrees, revealed that pitch and yaw errors of only 0.18 and 0.21 meters, respectively, would be expected.

4.63 The primary sources of lateral docking miss are range errors, residual thrust, accelerometer bias, and roll misalignment.[28] The contribution of each is listed below:

	Pitch	Yaw
1. Radar Errors	.16m	.16m
2. Residual Thrust	.051	.051
3. Accelerometer Bias	.054	.054
4. Roll Misalignment	0	.105
	.18m (rms)	.21m (rms)

Radar errors are due to errors in establishing angular rate and range errors; residual thrust errors are due to value and relay action errors (50 ms); accelerometer bias is due to a bias of .003 m/sec^2 (.01 fps^2); the roll error is due to 5° roll misalignment.

4.64 Table 9 analyzes the errors expected in docking as to source and magnitude.[29]

State of Arts:

4.65 In ranges of 5 to 25 meters, pulsed radar techniques can establish range to 1% accuracy. In range rates of 1 foot/sec. to 6000 feet/sec., pulsed radar techniques can establish range rate to 1% or $\pm$ 1 ft/sec. Angular rates of up to 114°/sec. can be established to $\pm$ 1 min/sec. Infrared horizon sensors and optical star trackers can currently supply attitude reference to within 0.5° accuracy; this will be improved to 0.1° in the next few years.

4.66 With little effort, optical techniques at ranges of 0 to 25 meters will be able to establish axial alignment to $\pm$ .06°, range to $\pm$ 0.1 foot, lateral displacement to 0.1 foot, and roll to 1.0°, respectively.

27/ Lockheed Aircraft Corp., op. cit.

28/ Lockheed Aircraft Corp., Missile and Space Division, Sunnyvale, Calif., Proposal for Orbital Docking Test Program, Volume I, Parts I and II, 26 June 1961, CONFIDENTIAL.

29/ Lockheed Aircraft Corp., op. cit.

TABLE 9
ERROR SOURCES AND UNCERTAINTIES FOR
ORBITAL DOCKING [1]

Type Error	Magnitude (1σ)
1. IRP Misalignment	
Roll and Pitch Axis Errors	
horizon sensor alignment	.033°
horizon sensor accuracy	.1°
equivalent horizon noise	.03°
thrust misalignment	.038°
control system errors	.03°
vehicle misalignment	.1°
	.156°
Yaw Axis Errors	
gyrocompassing error	—
thrust misalignment	.122°
control system error	.078°
vehicle misalignment	.1°
2. Gyro Drift Errors	
constant drift rate	.1743 deg/hr
mass unbalance	2.0 deg/hr/g
anisoelasticity	.02 deg/hr/g^2
3. Accelerometer Errors	
bias	4.658×10^{-5}g
scale factor	10.0×10^{-5}g/g
nonlinearity	$.45 \times 10^{-6}$g/g^2
misalignment	.033°
4. Ground Guidance System Errors	
Change in Tangential Velocity	.66 m/sec
Change in Radial Velocity	1.91 m/sec
Change in Normal Velocity	1.77 m/sec
Change in Tangential Position	7.22 ft
Change in Radial Position	57.4 ft
Change in Normal Position	42.6 ft

[1] Lockheed Aircraft Corp., Missile and Space Division, Sunnyvale, Calif., Proposal for Orbital Docking Test Program, Volume 1, Parts 1 and 2, 26 June 1961, [redacted].

4.67 Table 10 assesses the guidance state-of-the-arts and the requirements for orbital docking. It is evident that a technique for monitoring orbital docking and aligning the vehicles over the last few seconds and last few feet of travel will be required. Other than this, the requirements of guidance and the state-of-the-arts for orbital docking are compatible.

Orbital Transfer, Assembly, Repair, Maintenance, and/or Checkout:

4.68 For the scope of this study, guidance considerations in this mission are limited, and are dependent on the extent to which the operations are performed and on subsequent missions of the vehicles involved. Statements regarding the requirements of guidance during this mission are for the most part generalized.

4.69 Perturbations to the orbiting vehicle due to the performance of any part of this mission will have to be compensated in the guidance of the vehicles. For example, during external assembly, material transfer, or repair, the ephemeris of the vehicle could be altered since drag characteristics of the vehicle would be affected. Guidance will initiate corrective measures or establish the adjusted ephemeris to the accuracies desired, as discussed previously in the Launch and Orbit Mission. Perturbations due to internal factors, such as by the movement of equipment and personnel within the vehicle could arise, but these will be filtered out by the guidance sensors. It is not expected that perturbations of this type will be significant.

4.70 During the transfer of material or man between vehicles it is essential that the relative position of vehicles be maintained constant. Mating or direct contact of vehicles during this operation will resolve this problem. Without contact between vehicles, the ephemeris of the vehicles and the operational characteristics will have to be essentially identical—impossible using current guidance techniques and state-of-the-arts. Also, collision of the parallel orbiting vehicles is a danger that could occur under such mission conditions—the guidance of both vehicles will have to prevent this.

4.71 In orbital checkout, as in pre-launch checkout, the philosophy of checkout will be to stimulate the systems in the same manner and sequence as occurring in operation. Initial orbital checkout procedures will involve self checkout techniques with the results being telemetered to ground-support data receipt and processing stations. Later techniques will involve checkout between mated vehicles, i.e., orbital launch platform and a lunar vehicle. These checkout operations should establish the ability of the vehicles to maintain proper guidance during the current and subsequent missions.

TABLE 10

ANALYSIS OF DOCKING GUIDANCE REQUIREMENTS AND STATE-OF-THE-ARTS

Docking Requirements:

Parameters:

Range (max)	5 to 25 meters
Range Rate (max)	3 to 5 meters per second
Closing Period (max)	30 seconds
Lateral Displacement (max)	0.2 meter
Angular Alignment	1°

Measurement Accuracies:

Range	0.3 meter
Range Rate	0.3 meter/second
Roll Alignment	1.0° or better
Pitch Alignment	0.1°
Azimuth Alignment	0.1°

Docking State-of-the-Arts:

Measurement Accuracies:

Range (5 to 25 meters)	1%	
Range Rate	0.3 meter/second	
Pitch Alignment	0.06°	with little
Azimuth Alignment	0.06°	additional
Roll Alignment	0.2°	effort
Lateral Displacement	0.03 meter	

4.72 As far as can be determined, the effects of the mission flight conditions, which in turn define guidance requirements, have not been analyzed—it is considered negligible in comparison to other perturbations to the vehicle.

Earth Orbital Launch and Translunar Flight

4.73 Guidance responsibilities associated with this mission deal with: (a) launch or injection prediction and monitoring, (b) abort guidance, (c) pre-launch alignment of vehicle and guidance systems, (d) trajectory determination, (e) correction computation, (f) navigation, and (g) correction guidance.

4.74 Prior to orbital launch, it will be essential that the position, velocity and time relationships of the vehicle be established. This could be accomplished by vehicle sensors, or by ground support stations from which the information could be subsequently telemetered to the vehicle. Based on the vehicle and moon positions and ephemerides, and the position of the perilune or desired lunar landing corridor, a reference trajectory will be established. Optimum launch positions exist and will be established for each translunar flight depending on intent of the mission; an attempt should be made to launch from these positions. Anticipated errors and subsequent necessary corrections should be computed based on actual launch position, time, and velocity. When a trajectory is synthesized, the factors to be considered are:

1. Launch site or location.
2. Allowable launch azimuth.
3. Flight path angle.
4. Injection altitude.
5. Earth and moon ephemerides.
6. Desired lunar mission.
7. Flight time.

If the flight is to be cislunar or circumlunar, other considerations would be:

1. Return site.
2. Return route desired.

4.75 Primary sources of data for the determination of vehicle location and velocity will be the inertial platform, accelerometers, and the optical tracking systems. Backup information may come from ground support stations.

4.76 The accuracy requirements of the vehicle trajectory are very stringent. An inertial guidance system, whose errors are proportional with time, must be realigned at intervals using star or planet trackers which, in turn, have limited capability. Thus, consideration must be given to corrective maneuvers, and what must be done to keep these maneuvers at an economical level of propellant utilization. As in all guidance considerations, the requirements for such a mission are a function of the desired mission goals and correction capabilities.

4.77 A three dimensional lunar trajectory analysis established allowable errors for injection velocity, injection angle, heading angle and position angle for lunar impact and lunar orbit missions. This study assumed launch from a 300 statute mile circular earth orbit with no mid-course corrections. Tables 11 and 12 present data and a summation, respectively, of these errors. In the trajectory analyses, for each flight analyzed, only one error was assumed in order to establish its effect on the mission. However, combinations of errors in initial conditions were also investigated for several trajectories and it was found that miss distances were directly additive for the range of errors investigated. This study also established that "the accuracy required to hit within a hypothetical sphere with radius equal to the moon radius, but with center a few thousand miles from the center of the moon, is greater than that required to hit the moon. However, satisfactory orbits for a relatively close lunar satellite can be obtained with accuracies in initial conditions approximately equal to those required for lunar impact. When the space vehicle is spin-stablized at injection, careful consideration must be given to the choice of injection angle (and thus retro-rocket orientation) in order to achieve satisfactory lunar orbits." [30]

4.78 The penalty in the form of plane change associated with early or late launch is shown in Figure 12. [31]

4.79 A comparison of the results of investigations into allowable errors for orbital launch and translunar flight, and the guidance state-of-the-arts makes it evident that midcourse corrections are necessary. An analysis [32] resulted in recommendation of a total (3σ) midcourse correction of 93 feet per second and a reserve capability of 150 feet per second. A typical

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[30] Langley Research Center, Langley, Va., Three Dimensional Lunar Mission Studies, NASA Memorandum, June 1959, UNCLASSIFIED.

[31] NORAIR Division, Northrop Corp., Hawthorne, Calif., Preliminary Parametric Analysis for an Orbital Rendezvous Base System, 19 Oct. 1961, UNCLASSIFIED.

[32] North American Aviation, Inc., *op. cit.*

TABLE 11
ALLOWABLE ERRORS IN ORBITAL LAUNCH CONDITIONS FOR TRANSLUNAR FLIGHT

Injection Velocity Ratio	LUNAR IMPACT (ANYPLACE) Velocity (fps)	Injection Angle (degree)	Heading Angle (degree)	Postion Angle (degree)	LUNAR ORBIT RETRO POINT (5000 miles from Moon) Velocity (fps)	Injection Angle (degree)	Heading Angle (degree)	Position Angle (degree)	Nominal Flight Time (hrs)
0.992	(±10)	(+.55,-.75)	(+.75,-.80)	(+1.40,-1.50)	(±5)	(±.30)	(+.35,-.32)	(+.70,-.65)	73
0.995	(±25)	(±.40)	(±.60)	(±.75)	(±15)	(±.23)	(+.28,-.30)	(+.45,-.40)	58
0.998	(+50,-45)	(±.35)	(±.50)	(±.55)	(±30)	(±.20)	(+.25,-.28)	(+.40,-.37)	50
1.000	(+65,-60)	(±.30)	(±.48)	(+.30,-.25)	(±45)	(±.18)	(" ,-.27)	(±.35)	47
1.005	(+185,-100)	(±.27)	(±.43)	(+.26,-.20)	(+100,-85)	(±.15)	(" ,-.26)	"	42
1.006	(+190,-110)	(±.25)	(±.40)	(+.25,-.20)	(+105,-90)	"	(" ,-.25)	"	40
1.010	(+164,-140)	(±.23)	(±.38)	(+.24,-.20)	(+110,-110)	"	(±.25)		—
1.015	(+140,-160)	(±.20)	(±.38)	(+.23,-.20)	(+108,-125)	"	"	"	—
1.017	(+135,-165)	(±.18)	——	——	(+107,-120)	"	"	"	—
1.020	(+135,-160)	(±.18)	——	——	(+106,-110)	"	"	"	—
1.022	(+135,-145)	(±.18)	——	——	(+105,-100)	"	"	"	—

Note:
(a) This is taken from curves presented in NASA Memorandum, "Three Dimensional Lunar Mission Studies", June 1959.
(b) Injection Velocity ratio is the ratio of injection velocity to the normal velocity for injection of vehicle in 300 statue mile orbit (35,384.5 fps).
(c) Injection Angle is angle between velocity vector or thrust, and normal to radius.
(d) Heading angle is zero for due east, positive for north and negative for south of east firing.
(e) Position Angle is angle between radius vecotr at injection point and earth-moon axis. Errors in this angle are errors in firing time, 1° is approximately 4 minutes change in injection time.

[redacted]

TABLE 12

A SUMMARIZATION OF ALLOWABLE ERRORS IN ORBITAL LAUNCH CONDITIONS FOR TRANSLUNAR FLIGHT

PARAMETER	LUNAR IMPACT (Anyplace)	LUNAR ORBIT RETRO POINT (5000 miles from Moon)
Injection Velocity	(± 15 fps) to (-165, + 190 fps)	(± 10 fps) to (-125, + 110 fps)
Injection Angle	(± 0.5°) to (± 0.1°)	(± 0.3°) to (± 0.1°)
Heading Angle	(± 0.75°) to (± 0.4°)	(± 0.35°) to (± 0.2°)
Position Angle	(± 1.5°) to (± 0.5°)	(± 0.75°) to (± 0.25°)

(1) This is from NASA Memorandum, "Three Dimensional Lunar Mission Studies", June 1959.

(2) Position angle is position with respect to earth. Position Angle of 1° is equal to approximately 4 minutes of time.

(3) Allowable errors are a function of injection velocity.

(4) Study assumed a 300 statute mile earth circular orbit for launch.

[redacted]

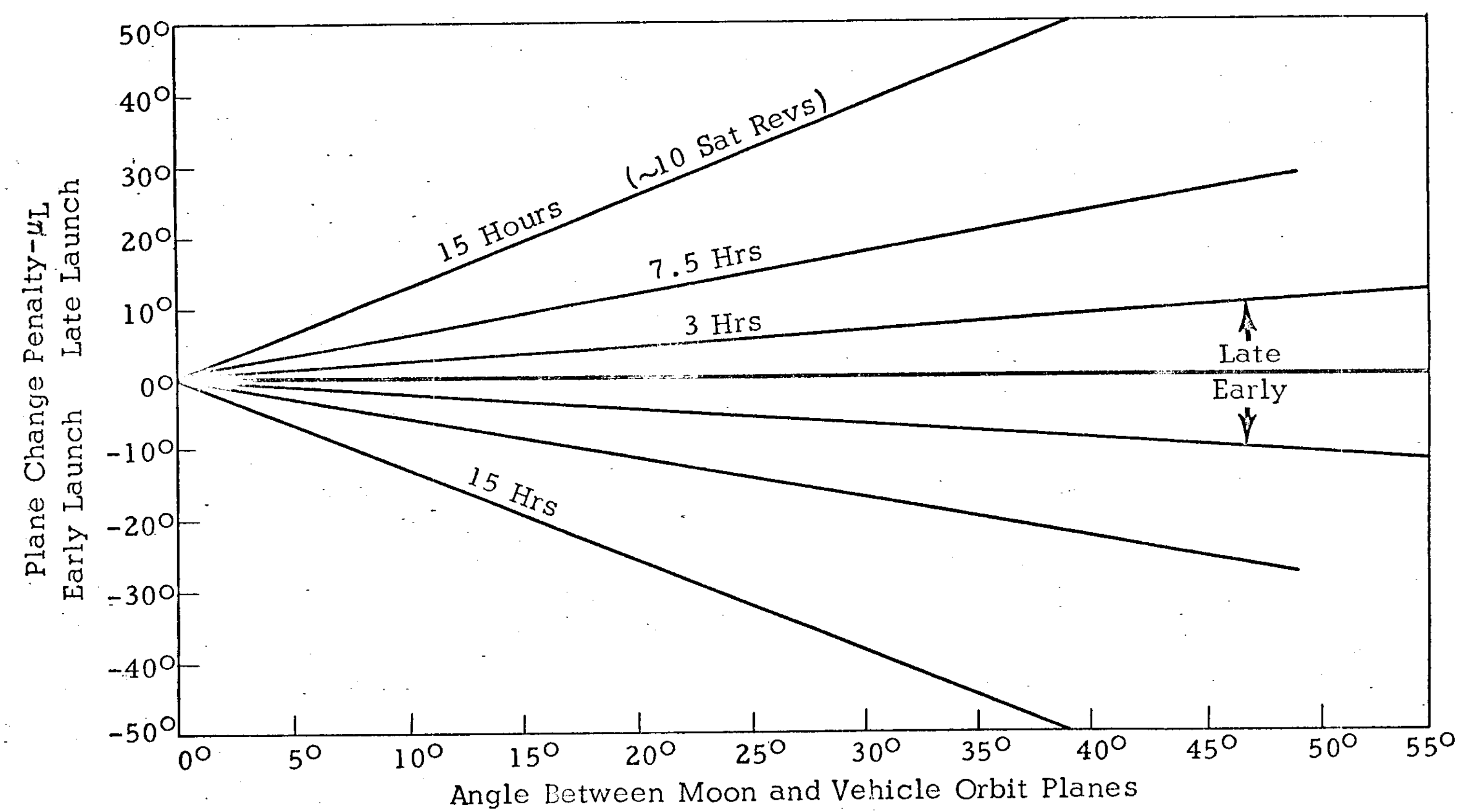

FIGURE 12. EARLY/LATE LAUNCH PENALTY

precision accelerometer could give a total velocity error of 11 feet per second in a 3 hour injection; a midcourse correction of about 90 feet per second will be required. A drift of 0.06 degree in 3 hour lift-off will require a correction of 12 feet per second to avoid a 250 statute mile miss of destination.

4.80 Centaur guidance could establish characteristics of the launch vehicle translunar flight such that the following midcourse corrections would be as indicated:[33]

1. Orbital Launch:

   Position, σ = 2 n.m.

   Velocity, σ = 4 fps

   Flight Path, σ = $.02^{\circ}$

2. Midcourse correction 145,000 nm from earth:

   Velocity correction, σ = 4.5 fps (earth direction)

3. Midcourse correction 17,000 miles from moon:

   Velocity correction, σ = 3.0 fps (each component)

4. Perilune 67 miles from moon:

   Position, σ = 1 n. m. (each component)

   Velocity, σ = 2.0 fps (each component)

5. Earth Re-Entry:

   Position, σ = 6 n.m. (horizontal)

   σ = 0.1 n.m. (altitude)

   Velocity, σ = 7 fps

   Flight Path, σ = $.02^{\circ}$

4.81 This is based on a mathematical trajectory analyses of the translunar and transearth flight assuming the following system capabilities:

Star Tracker:	σ = 1/6 minute
Horizon Sensor:	σ = 6 minutes
Planet Tracker:	σ = 1/3 - 1 minute

[33] NASA, NASA-Industry Apollo Technical Conference-A Compilation of the Papers Presented July 18, 19, 20, 1961, [redacted]

Gyro-Stable Platform:

Fixed Drift	3σ	$= 0.05^{\circ}/hr.$
Mass Unbalance	3σ	$= 0.1^{\circ}/hr/g$
Anisoelastic	3σ	$= 0.02/hr/g^2$

Accelerometers:

Bias	3σ	$= .0001\ g$
Scale Factor	3σ	$= .0001\ g/g$

This apparently is representative of the near future state-of-the-arts. However, the star and planet trackers, horizon sensor and stable platform performances seem optimistic.

4.82 Table 13 lists the translunar guidance requirements for a number of lunar missions[34] Figure 13 shows the sensitivity of the perilune attitude to vehicle launch velocity, flight path angle and time error of launch as a function of burn-out velocity.

4.83 It is obvious that the state-of-the-arts of guidance is not adequate to fulfill the requirements of translunar flight, without midcourse corrections. The anticipated amount of corrections necessary seems to vary; however, the concensus of opinion suggests up to a total of 250 feet per second. This could be very optimistic. With such a corrective capability, current guidance techniques and capabilities are supposedly adequate for any of the translunar flight missions: cislunar flight, circumlunar flight, lunar orbit, or lunar land.

4.84 There is a definite need for a thorough analysis of corrective requirements for translunar flight before the development of the launch vehicle is begun. Such information as magnitude, frequency, time, place, scheduling, and the basic correction philosophy should be established as a function of error, reliability, and confidence level.

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[34] H. A. Lieske, Rand Corp., *op. cit.*

[redacted]

TABLE 13

EARTH ORBITAL LUNAR LAUNCH GUIDANCE ACCURACY REQUIREMENTS

LAUNCH VEHICLE MISSION	REQUIREMENTS (3σ) Velocity Tolerance (fps)		Angular Tolerance (degrees)
a. Cislunar Flight:			
1. to ± 100 mn of desired distance, return to earth (anyplace)	1.0	or	0.01
b. Circulunar Flight:			
1. Perilune ± 40000 nm, return to earth (anyplace)	150	or	10
2. Perilune ± 40000 nm, return to earth ± 50,000 perigee	1.0	or	0.001
3. Perilune ± 40000 nm, return to earth ± 1000 nm	0.25	or	0.03
4. Perilune ± 15000 nm, return to earth (anyplace)	150	or	2.5
5. Perilune ± 100 nm, return to earth (anyplace)	4	or	.02
6. Perilune ± 30 nm, return to earth (anyplace)	1	or	.005
c. Lunar Orbit			
1. to ± 100 nm of desired lunar orbit	4.0	or	0.02
d. Lunar Land			
1. Hard impact, surface toward earth (anyplace)	75.0	or	0.5
2. Hard impact, ± 100 nm of desired place	4.0	or	0.02
3. Hard impact, 1 nm of desired place	0.3	or	0.0001

[redacted]

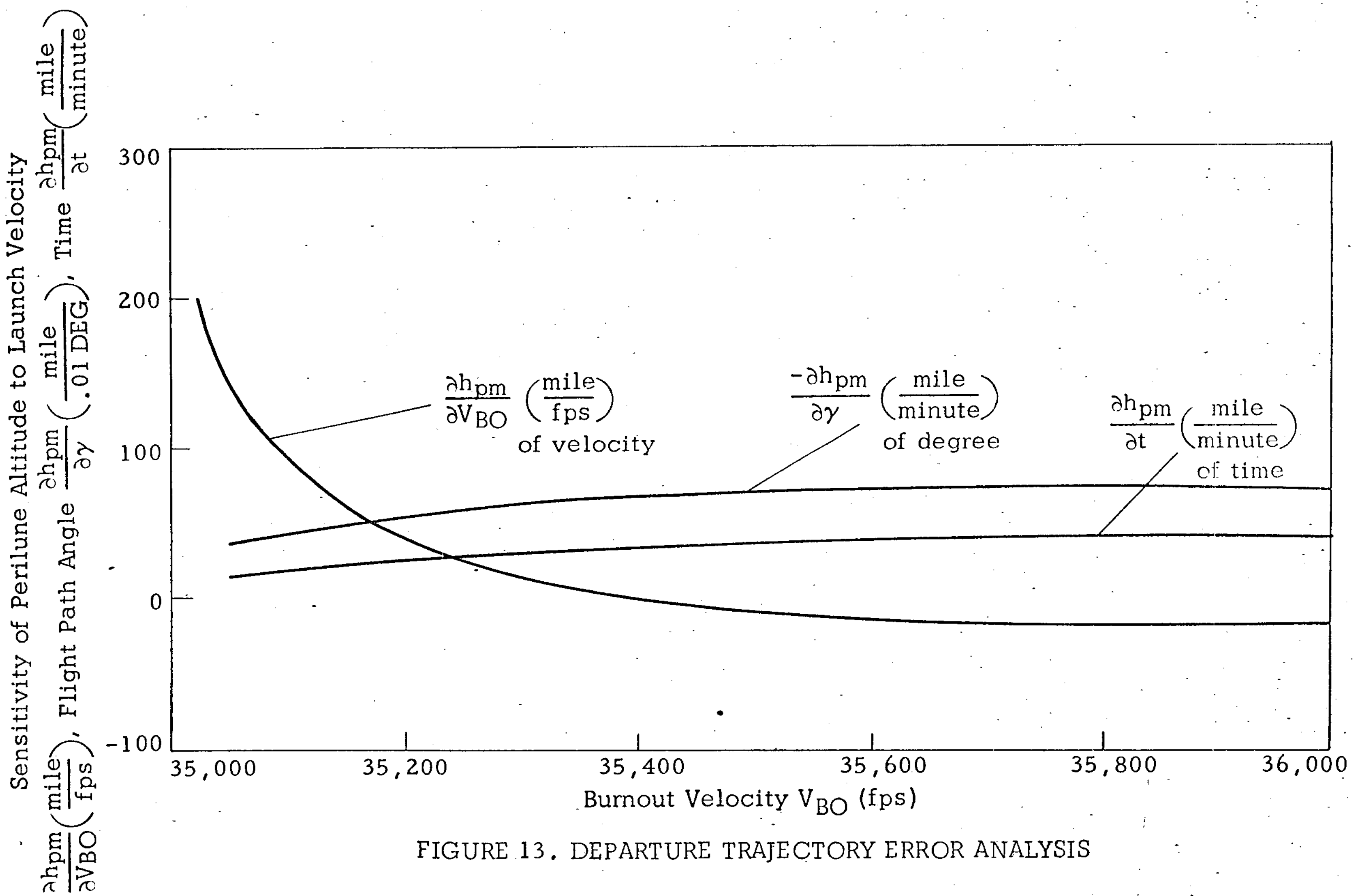

FIGURE 13. DEPARTURE TRAJECTORY ERROR ANALYSIS

Lunar Orbit and Landing

4.85 The velocity of the vehicle as it approaches the moon depends on the velocity of escape from the earth; it will probably be slightly greater than 9000 feet per second. As it approaches the desired lunar entry corridor, it will retrograde into orbit at a velocity approaching 6000 feet per second. Depending on the magnitude of the retrograde impulse, the trajectory will be elliptic (or circular), parabolic, or hyperbolic. The elliptic orbits are required for survey or lunar land purposes.

4.86 Perturbations in lunar orbit will be due to the earth's gravity field, the sun's gravity field, the moon's potential distribution and lunar librations. For short duration low lunar altitude orbiting, the effects of these perturbations are not significant.[35]

4.87 The functions of lunar landing guidance would be to: (1) establish and monitor vehicle position with respect to the landing site, (2) establish and monitor vehicle velocity, (3) establish and monitor retro thrust to assure proper change in velocity, (4) establish, monitor and maintain proper orientation of space craft, (5) monitor rate of descent, and (6) establish correct orbit plane.

4.88 The Lunar Landing technique desired for the Apollo vehicle has been analyzed;[36,37] the predicted time history for lunar landing of the vehicle is presented in Figure 14. From the translunar trajectory traveling at a 9000 fps, the vehicle will retro into a 100 statute mile circular orbit (period of orbit = 2.1 hours) with a Δ V of 3025 fps, survey the landing area while in orbit, then when the spacecraft is 180° from the landing site, retro with a Δ V of 180 fps into elliptic orbit with 50,000 ft pericynthion. It is anticipated that lunar landing from this altitude will take nominally 375 seconds, during which the velocity decreases from 6000 fps to zero

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[35] Morris V. Jenkins and Robert E. Munford, Preliminary Survey of Retrograde Velocities for Insertion into Low Altitude Lunar Orbits, NASA TN D-1081, September 1961.

[36] NASA-Industry Apollo Tech. [redacted], op. cit.

[37] NASA, Langley Field, Virginia, Project Apollo Spacecraft Development Statement of Work, Phase A, Request for Proposal No. 9-150, Project Apollo Spacecraft, 28 July 1961, [redacted]

[redacted]

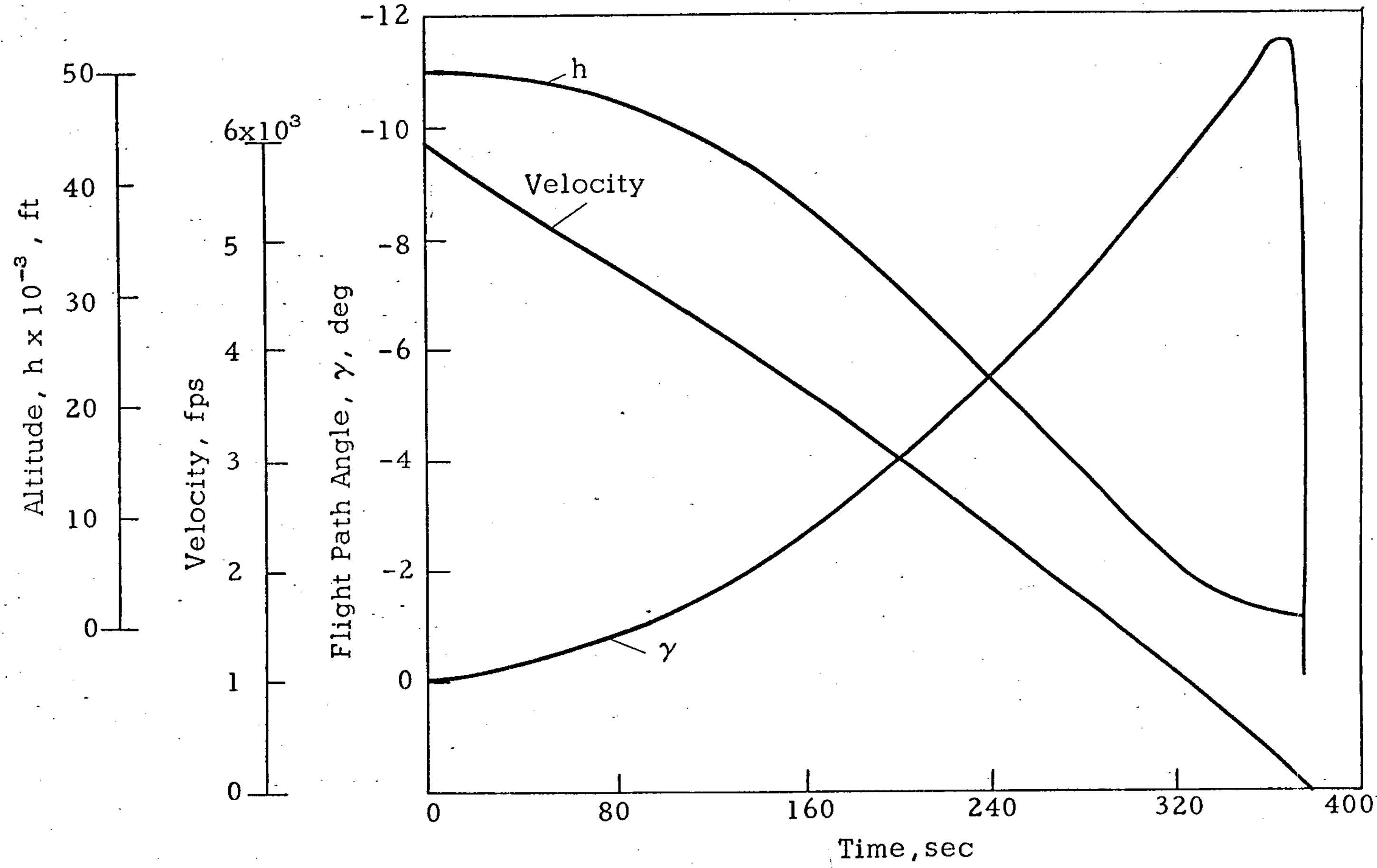

FIGURE 14. TIME HISTORY FOR A LUNAR LANDING FROM AN ELLIPTICAL ORBIT OF PERIGEE ALTITUDE OF 50,000 FT FOR APOLLO

[redacted]

at a rate of approximately 1000 fps/minute. The maneuver ends at an hover altitude of 100 feet with vertical velocity at zero and horizontal velocity at less than 25 ft/sec. From hover, the vehicle would touch down at a very slow rate, in the orientation desired. The change in velocity on impact should not exceed 1 fps.[38]

4.89 As far as can be determined, no detailed analysis of guidance requirements for this mission has been conducted. Only statements are presented which indicate that the vehicle can be guided through the mission outlined. One study indicated–"The guidance of a vehicle to a soft landing on the moon has been examined. It is concluded that guidance in two stages, lateral guidance at a point some 5,000 miles above the surface using optical and inertial instruments as error sensors, and braking along the path near the surface of the moon using a radio altimeter as an error sensor, is feasible. Existing components and techniques are adequate to perform these functions; aside from weight reduction, no new developments are needed.[39]

4.90 However, based on engineering judgment, to keep corrections at a minimum it seems that the horizon should be established to 0.5°, velocity to 1%, velocity change to 1%, altitude to 1%, attitude change to 1%, and orbit plane to 1°. These estimates are essentially "ball park" values; they do not reflect an analysis of the type that should be conducted.

Lunar Launch and Transearth Flight

4.91 The functions of the guidance system during this mission will be the same as those outlined previously in the discussions of the Earth Launch and Orbit Mission, and the Earth Orbital Launch and Translunar Flight Mission. During lunar launch, ascent and orbit, the guidance system will compute and monitor velocity, position, and time; the accelerometers will monitor vehicle velocity and the inertial stable platform will monitor orientation of the vehicle.

4.92 During transearth flight, guidance will deal with injection prediction and monitoring, abort guidance, trajectory determination, correction computation, navigation, and correction guidance. Injection into transearth flight

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[38] NASA, Langley Field, Va., Request for Proposal No. 9-150, op. cit.

[39] C. R. Gates, Terminal Guidance of a Lunar Probe, Jet Propulsion Laboratory, External Publication No. 506, May 14, 1958.

can occur from a cislunar or a circumlunar flight, on direct launch from the moon, or from lunar orbit; the latter technique is to be used in the Apollo vehicle mission. When the route to earth is to be determined, factors to be considered are:

1. Launch location
2. Launch azimuth
3. Flight path angles
4. Inspection altitudes
5. Earth and moon ephemerides
6. Flight time
7. Return site
8. Type of re-entry vehicle.

Again, the primary sources of information will be the inertial stable platform, the accelerometers, and optical tracking systems, and possibly earth support systems. Since this is another long-time mission, correction capabilities are very important considerations to be made in this mission.

4.93 On a direct flight from moon to earth, the lunar launch will have to be coordinated carefully with the earth ephemeris information; this may have to be forwarded to the vehicle prior to its launch. Command link problems will be considered in the Technical Area of Communications. Lunar orbit launch presents the vehicle a greater launch window for the transient flight than that of moon surface launch. At this time, it is not known if a study of the surface launch versus orbit launch has been conducted.

4.94 Figure 15 shows the predicted time history for the lunar launch of Apollo to a 100,000 foot parking orbit.[40] The vehicle will be launched vertically into the parking orbit at a velocity of 6,000 feet per second. At the appropriate time, transearth flight will be accomplished by a velocity increment of 3110 feet per second.

4.95 A return trajectory error analysis,[41] established the velocity and lunar launch angle extremes for successful transearth flights. There is a total spread allowance of over 800 feet per second in lunar burnout velocity over a wide band (-20° to 60°) of launch angles for return to earth by direct or retrograde route. This same study indicated the sensitivity of the perigee altitude to various initial parameters: path angle, launch position, azimuth angle and injection velocity. The analysis results, shown in Figures 16, 17, and 18 , indicated that the more desirable launch position angles would be in a range of 40° or greater, the range at which the effects of errors are minimized.

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[40] NASA, Langley Field, Va., Request for Proposal No. 9-150, op. cit.

[41] NORAIR Div., Northrop Corp., op. cit.

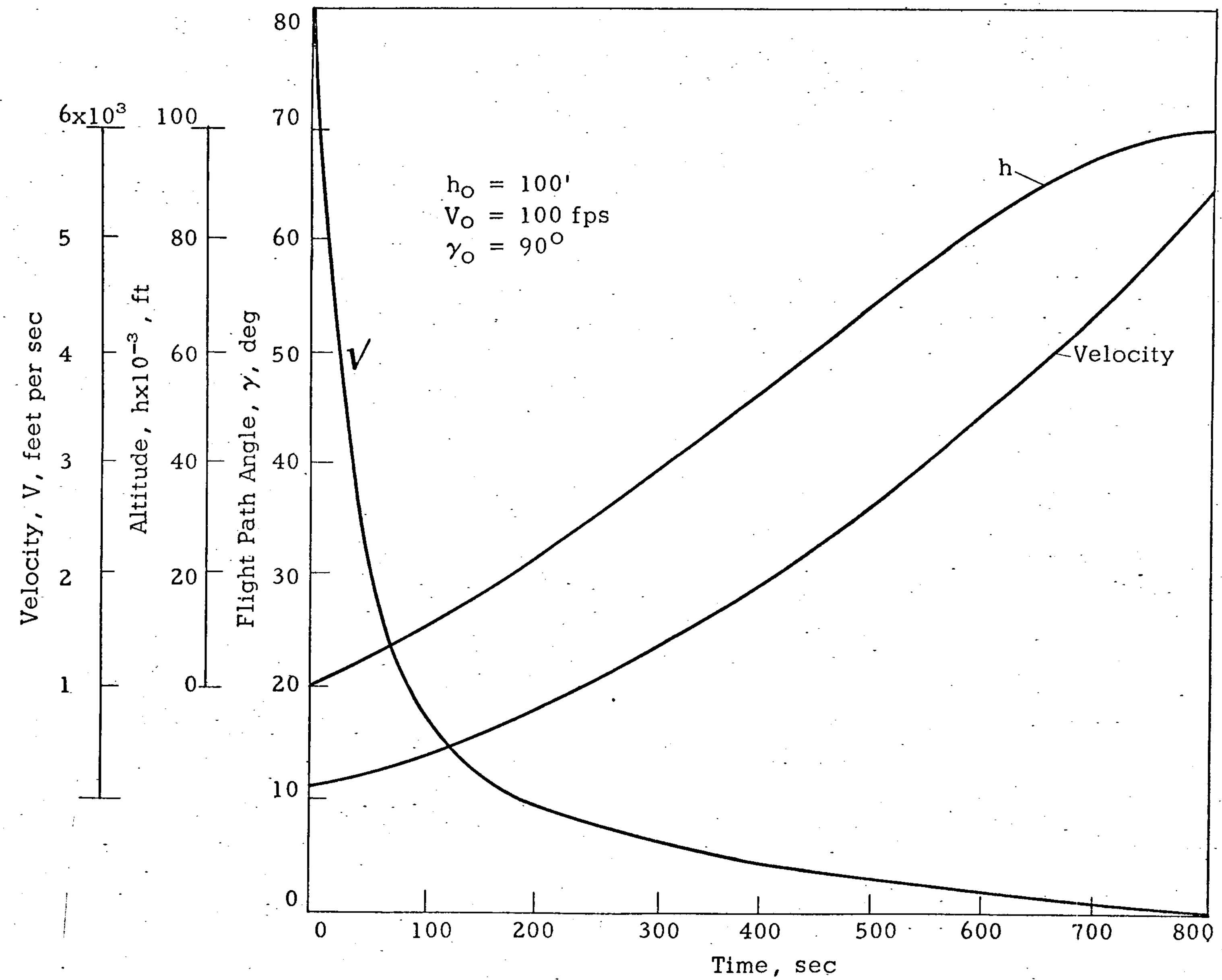

FIGURE 15. TIME HISTORY FOR A LUNAR TAKE-OFF TO A 100,000 FT PARKING ORBIT

[redacted]

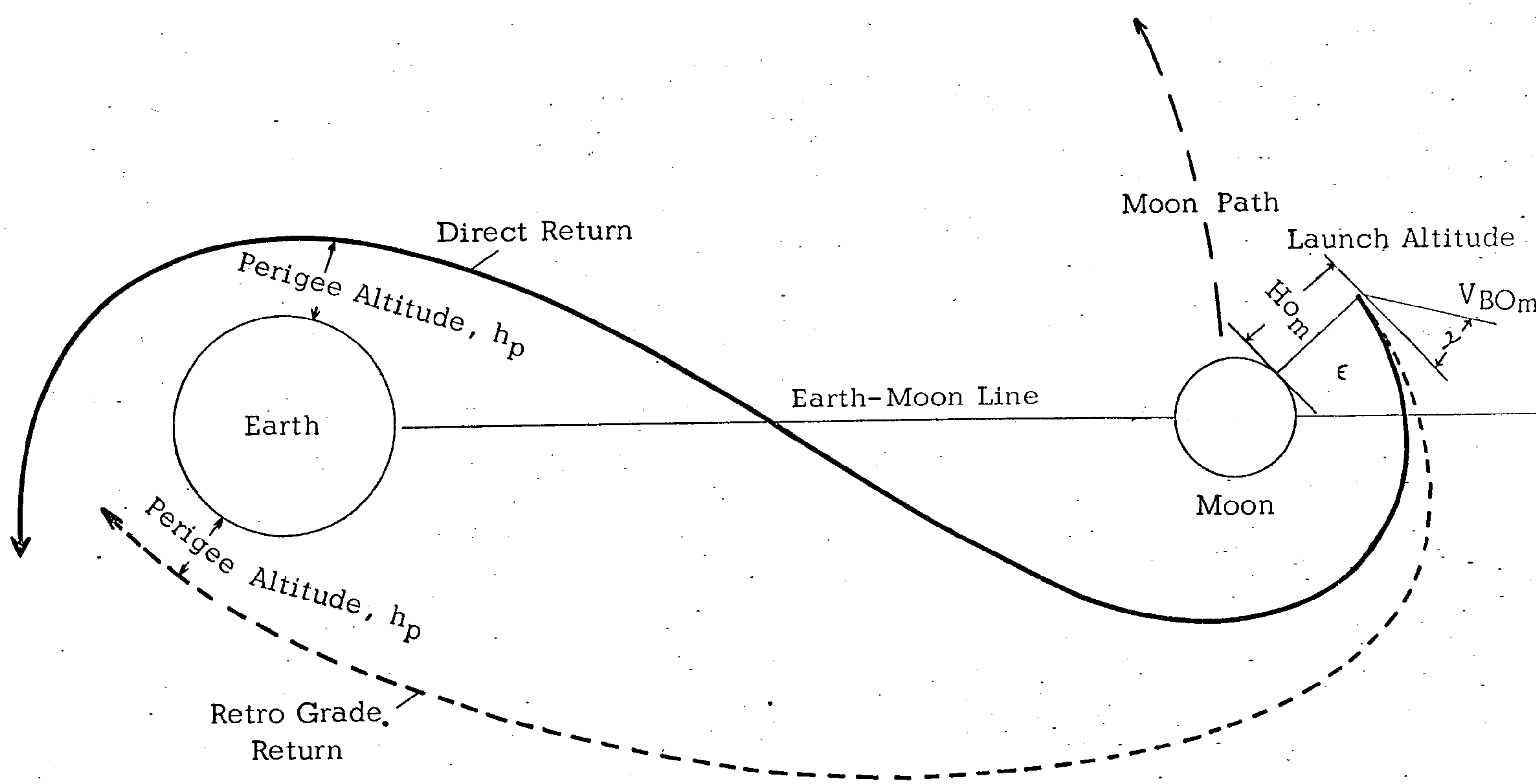

FIGURE 16. RETURN TRAJECTORY PARAMETERS

[redacted]

[redacted]

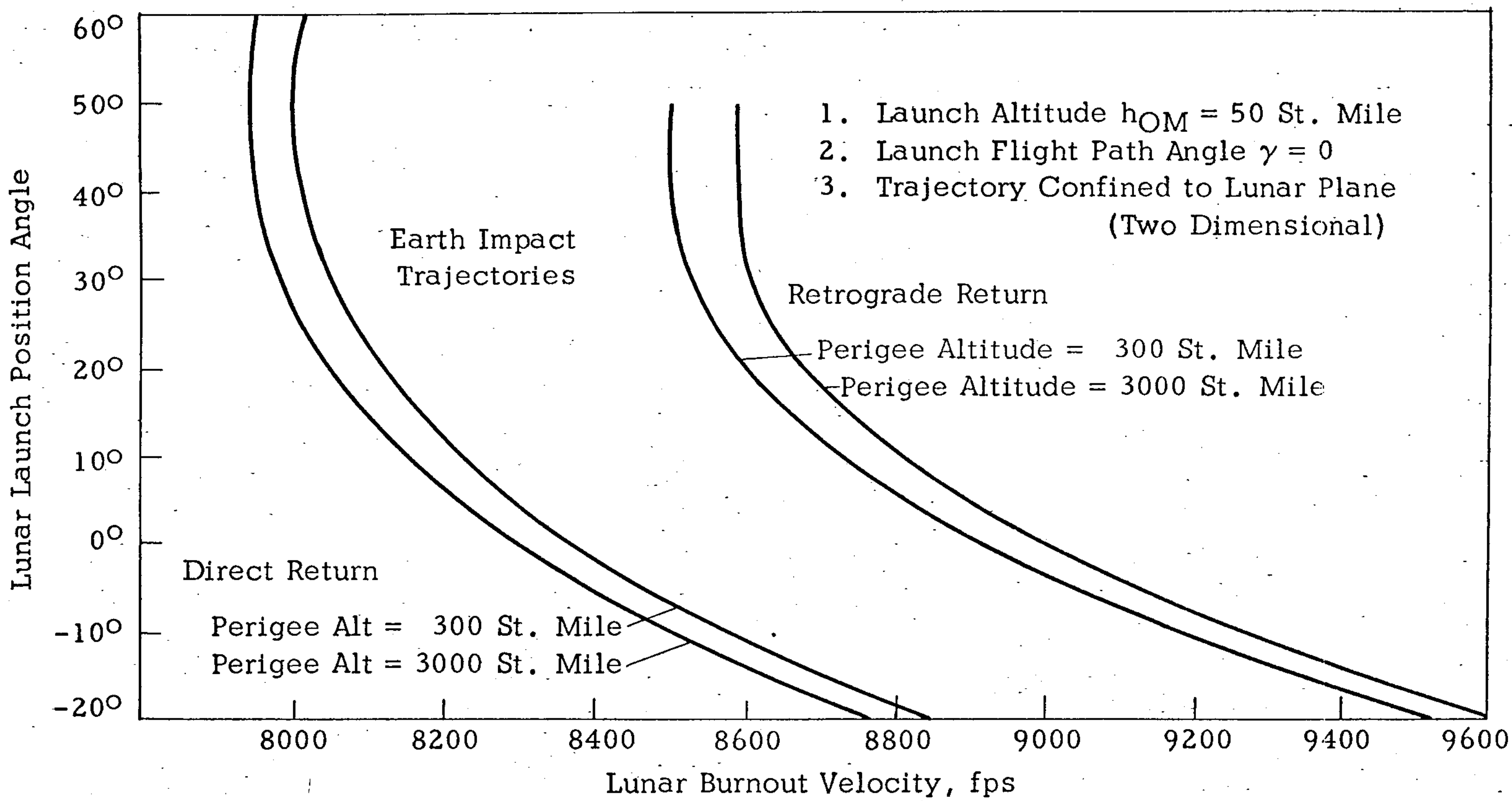

FIGURE 17. DISPLAY OF RETURN TRAJECTORY PARAMETERS

[redacted]

[redacted]

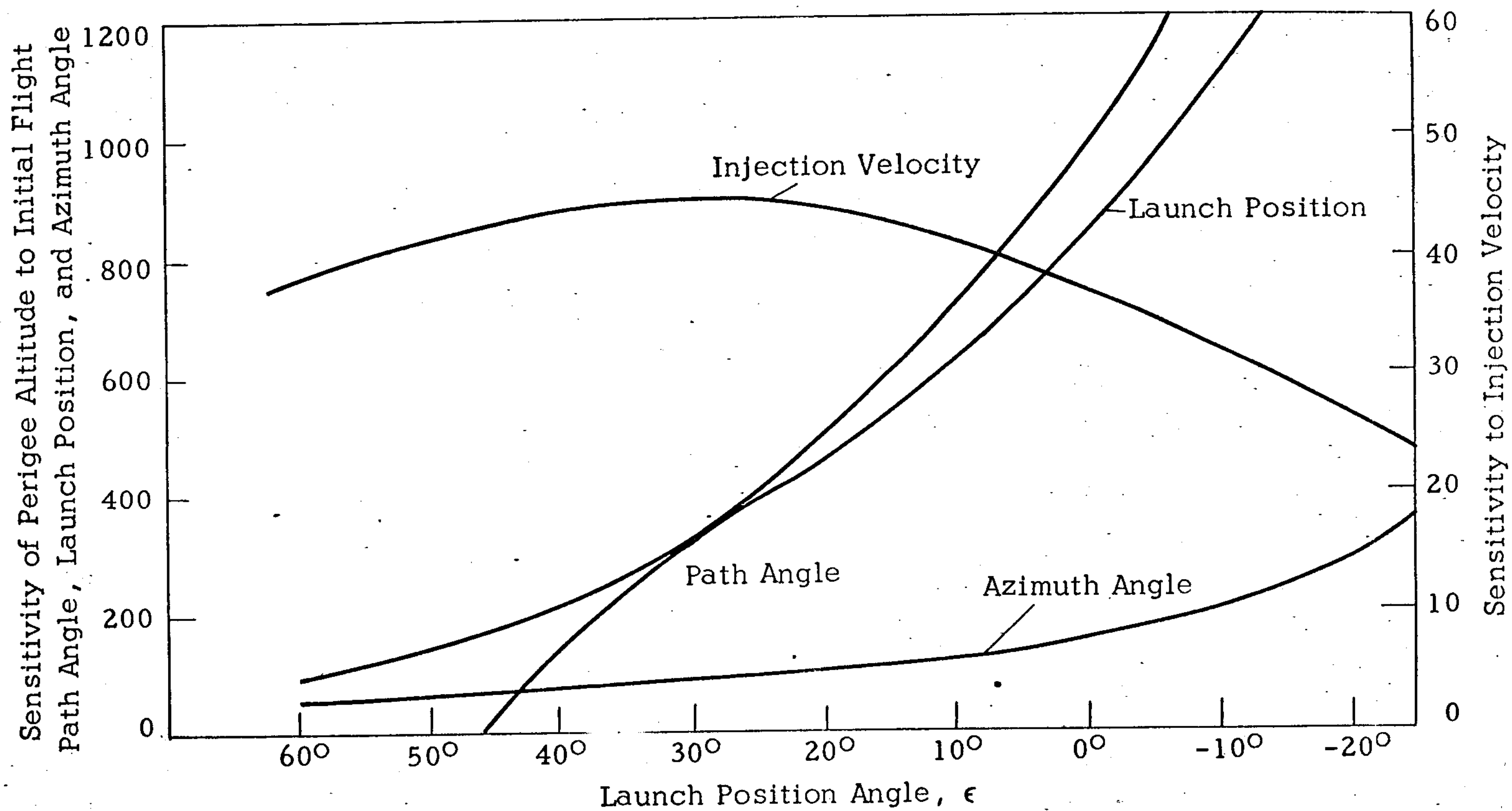

FIGURE 18. RETURN TRAJECTORY ERROR ANALYSIS

[redacted]

4.96 According to Table 14, for a launch angle of 60°, if a perigee error of no more than 10 miles was desired, the launch position error could not exceed 0.1°, <u>or</u> the azimuth angle error could not exceed 0.2°, <u>or</u> the injection velocity error could not exceed 0.25 fps. Since each of these errors is just within or exceeds the current state-of-the-arts, it is evident that midcourse correction will be necessary.

TABLE 14

EFFECT OF LUNAR LAUNCH ERRORS ON PERIGEE

Types of Errors	Lunar Launch Angles			
	0°	20°	40°	60°
Path Angle	980 mi/deg	500 mi/deg	130 mi/deg	0
Launch Position Angle	840 mi/deg	480 mi/deg	220 mi/deg	100 mi/deg
Azimuth Angle	175 mi/deg	100 mi/deg	70 mi/deg	50 mi/deg
Injection Velocity	37 mi/fps	44 mi/fps	44 mi/fps	38 mi/fps

4.97 There seems to have been very little effort directed toward analyzing the optimum time, place and magnitude of midcourse corrections. A study by NASA investigated the effects of random errors in velocity and flight path angle on the guidance correction of a space vehicle approaching the earth.[42] This showed that (1) velocity needed for correction maneuvering was less for most cases when a planned correction time or place was utilized, as compared to relying on a deadband control. Better perigee control was achieved when the deadband was omitted; (2) correction with deadband limit was more sensitive to initial conditions, instrumentation inaccuracies, location of final correction point, and degree of confidence required than a correction without a deadband limit; (3) if a deadband was used, it is more efficient to correct to the nearest boundary of the deadband than to the center of the deadband if the shift in the perigee altitude can be tolerated; (4) a deadband based on 3 times the standard deviations of the errors in the space vehicle velocity and flight path angle would not be satisfactory because of the high corrective thrust requirement and poor perigee-altitude control.

4.98 Additional similar analysis of transearth phase of the manned lunar mission will have to be conducted in order to ascertain the midcourse correction scheduling and magnitude throughout the flight.

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[42] Jack A. White, <u>A Study of the Effect of Errors in Measurement of Velocity and Flight-Path Angle on the Guidance of a Space Vehicle Approaching the Earth</u>, NASA TN D-957, October 1961, UNCLASSIFIED.

4.99 On the transearth flight,[43] velocity increments need for correction may exceed 7000 feet per second depending on the guidance scheme utilized and the correction format assumed. This is much greater than the nominal 250 feet per second suggested in other studies.[44,45,46] This difference reveals the necessity of further study in this area for corroboration of results and optimization of corrective maneuvers.

Earth Re-entry and Land

4.100 Guidance of the space-craft in this mission will assure that the vehicle utilizes the most appropriate landing corridor and techniques for successful landing at a predetermined destination. Factors affecting the mission technique and the guidance functions, are

a. re-entry vehicle design (aerodynamic properties such as W/C_DA and L/D).

b. availability of propulsion for controlled re-entry, atmospheric flight, retro, attitude control, etc.

c. respective positions of earth, spacecraft and the destination at re-entry.

d. atmospheric conditions and environments tolerable

e. accuracy with which re-entry corridor can be attained

f. retro technique

4.10 Prior to and during re-entry, flight guidance parameters to be determined and monitored are:

a. attitude

b. rate of attitude change

c. velocity

d. rate of velocity change

e. re-entry orbit plane (inclination to earth equator).

f. ground range to impact

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[43] Alan L. Friedlander and David Harry, III, An Exploratory Statistical Analysis of a Planet Approach Phase Guidance Scheme Using Angular Measurements with Significant Error, NASA TN D-471, September, 1960.

[44] NASA, Huntsville, Alabama, op. cit.

[45] General Dynamics/Astronautics, op. cit.

[46] North American Aviation, op. cit.

g. latitude of re-entry beginning

h. horizon (for attitude control).

i. re-entry angle.

These parameters should ultimately be determined within the vehicle system with backup from ground support systems via the command link. Communication between ground and the vehicle could be interrupted in the altitude range of 150,000 to 350,000 feet by the ion belts.

4.102 The Apollo flight plan calls for a perigee of 120,000 feet, velocity at perigee of 36,320 ft/second, and a reference re-entry altitude of 400,000 feet. The spacecraft characteristics during re-entry are L/D of 0.5 and W/C_DA of 50. The vehicle will use parachutes for slowdown; the specified vertical velocity calls for a maximum of 30 fps at a 5,000 feet altitude. See Figure 19.[47]

4.103 At the time of publishing of this report, the guidance requirements for earth reentry and land had not been ascertained.

47/ NASA, Langley Field, Virginia, Request for Proposal No. 9-150, op.cit.

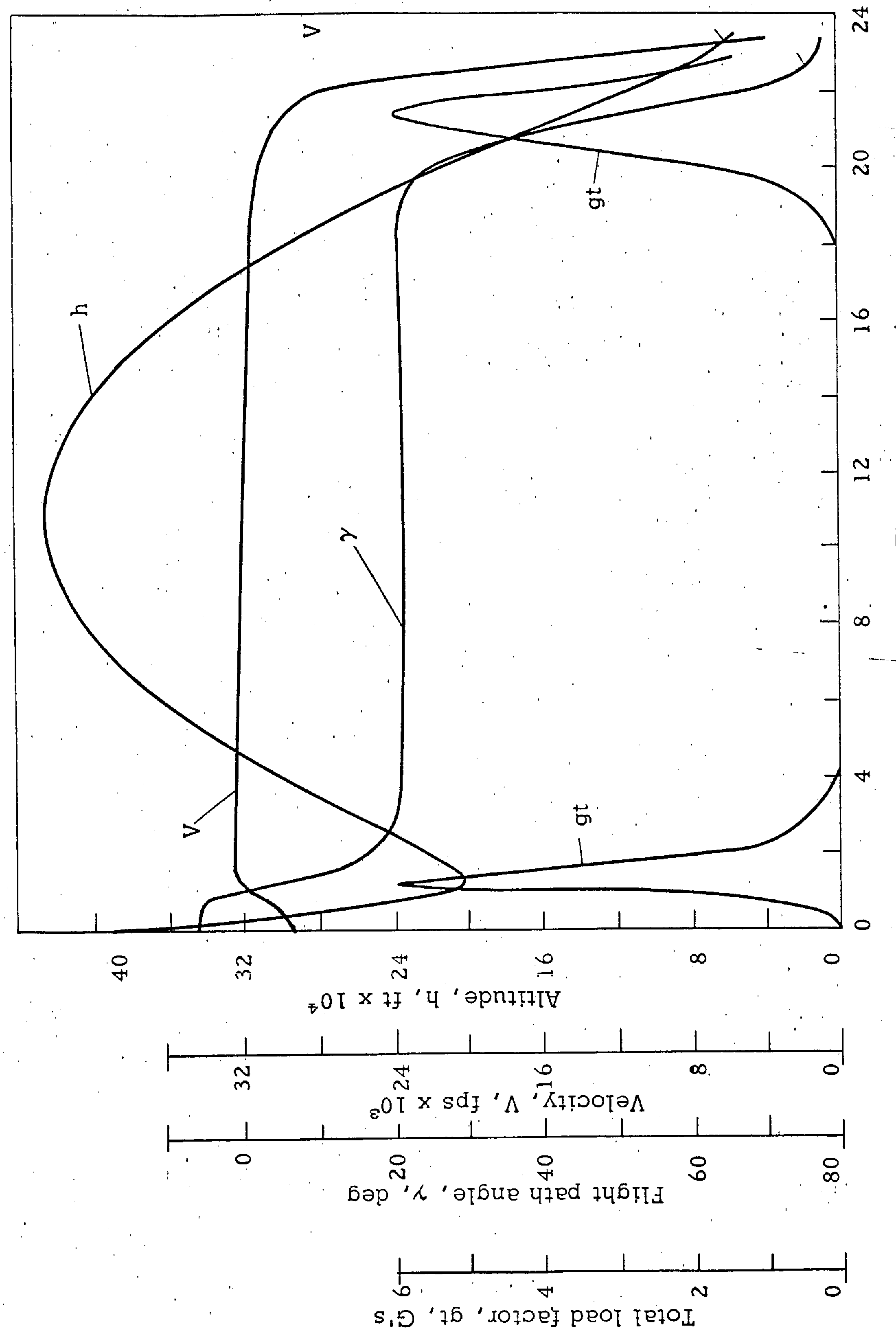

FIGURE 19. TIME HISTORY FROM REENTRY TO NEAR-LANDING FOR APOLLO

INVESTIGATIONS

4.104 This assessment of guidance for the various lunar missions has made it evident that there are investigations, analyses, and improvements in the state-of-the-art that will have to be attained to insure successful lunar mission flights. Some of the investigations are pertinent to specific missions; others are pertinent to all of the missions. These latter considerations include generalized improvements in guidance techniques and system performance, design, and manufacture.

4.105 NASA sponsored studies conducted by various contractors have indicated requirements for a number of guidance investigations. A few of these investigations have already been incorporated into the NASA Technology Support Program of the manned lunar mission. These are:

Investigations recommended by Lockheed Missile Systems Division pertinent to earth orbital docking:

1. "Investigate need for star trackers since it represents sizable step in complexity of system. The question of its necessity is directly related to mission error sensitivities and to the correction capabilities of the midcourse guidance and propulsion systems of the payload. It will be duplicating guidance system capabilities. Improvement over a horizon sensor/gyrocompass system would be equivalent to an error in velocity at lunar injection of 15-25 fps."
2. "Consider dual guidance operations in order to achieve a high probability of completing mission. The interconnection and change-over problems must be considered."

An investigation recommended by North American, Space and Information Systems Division pertinent to translunar flight:

1. "Analyze guidance accuracies associated with lunar flight trajectories of various energies, azimuth angles and injection angles with a breakdown of the criticalness of each error and a corrective maneuver."

An investigation recommended by Vought Astronautics pertinent to earth orbital launch:

1. "Conduct a study to evaluate the interaction of orbital launch point tolerances (exit gate) of orbital launch vehicles for each of the vehicle missions developed for the spectrum of orbital launch facilities involved.

Investigations recommended by General Dynamics, Astronautics:

1. "A rendezvous error study is necessary to explicitly define the mission, and establish guidance and control components and specifications. Three categories of errors should be considered. These are the mission, the error source, and error accumulation. Typical error sources are: Launching into orbit tracking within orbit; and maneuvering thrust errors. A co-variance matrix of position and velocity should be used for each mission segment."

2. "Establish the trade-offs between vehicle and ground based tracking must be made. The effect of orbit altitude on the number and location of sights to establish ephemeris must be determined. Command and control concepts and procedures for each segment of the mission must be established.

3. "For orbital docking, fluid motion in vehicles must be simulated. NPSH requirements in the zero "g" field must be established. Translational acceleration history and stochastic description of torque control history must be determined. Sensor accuracy and misalignment tolerance must be established. Bumping loads must be analyzed by defining probable points of contact and relative velocities."

4. "The manned lunar landing and return by an orbital launch boost system is an extremely complex four dimensional mission. Various segments of the problem have been extensively analyzed in two or three dimensions. Very little has been done to simulate the entire mission in four dimensions, where the fourth dimension is time. Time is all important in determining optimum mission procedures.

Since an infinite number of chronological mission sequences could occur to launch and guidance dispersions and system reliabilities, a statistical simulation of random failures and dispersions should be made. Such a simulation would include the spectrum of possibilities that must be analyzed in determining the operational procedures and requirements.".

5. "Operational testing and evaluation of the booster system and its associated rendezvous technique could conceivably represent forty percent of the total program cost. This economic consideration necessitates a detailed analysis of the effect of mission and operational requirements on the test and evaluation program. Unnecessary operational requirements can result in considerable time and cost extension of the test program. Such requirements should be determined and modified at the earliest possible date."

Investigations recommended by NASA:

1. Study of tracking facilities and accuracies associated with orbital launch vehicle to determine how accurate tracking must be to be consistent with docking alignment.
2. Development of a closed loop air supply system for recycling air in platform system—emphasize efficiency.
3. Investigate servo-system of accelerometer transducer, torques and position sensors.
4. Microminaturization of electronics with possibility of active filtering networks.
5. Investigate spin motor bearings for long life under varying environmental conditions.
6. Investigate accelerometer design error budget with objective of reducing threshold "g" and increasing dynamic range.
7. Minimize vibration and temperature transmission ability of gimballing materials and bearings.

4.106 Some of these investigations are not as broad in scope as they need to be, others are definite prerequisits to successful manned lunar missions.

4.107 Table 15 outlines the studies, analyses, and research and development investigations recommended by ORI to be essential contributions to the successful guidance of manned lunar vehicles. This table also indicates mission applicability; thus, if a mission is omitted from NASA plans, the effected tasks could be disregarded. These investigations are considered further in the technical area plans.

TABLE 15. RECOMMENDED PROGRAM SUPPORT INVESTIGATIONS

APPLICABILITY	TYPE OF TECHNOLOGY SUPPORT	
(SECONDARY MISSIONS)	STUDIES AND ANALYSES	RESEARCH AND DEVELOPMENT
EARTH ORBIT AND LAUNCH	None	None
ORBITAL RENDEZVOUS	1. Analyze the requirements and capabilities of ground support toward accomplishing unmanned/manned orbital rendezvous. 2. Conduct a rendezvous error analysis.	1. Improve the earth horizon sensor performance. 2. Improve altimeter performance: continue investigation of CW Doppler technique.
ORBITAL DOCKING	1. Analyze and establish guidance technique and philosophy for mated assembly. 2. Analyze manned versus unmanned docking guidance capabilities.	1. Develop short range (0-10ft) range and range rate techniques and equipment. 2. Improve optical alignment techniques. 3. Investigate television techniques for guidance. 4. Improve command link techniques for unmanned vehicle docking.
ORBITAL TRANSFER, ASSEMBLY, REPAIR, MAINTENANCE, AND/OR CHECKOUT	1. Analyze effect of mission on vehicle ephemeris, and establish corrective capabilities required. 2. Analyze the effect of the ephemerides of undocked vehicles on the transfer of material between vehicles. 3. Establish orbital and in-flight repair, test and maintenance philosophy for guidance.	

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(Continued)

TABLE 15. RECOMMENDED PROGRAM SUPPORT INVESTIGATIONS

EARTH ORBITAL LAUNCH AND TRANS-LUNAR FLIGHT	1. Analyze midcourse corrections to establish optimum time, place, and magnitude of correction parameters. 2. Analyze guidance accuracies associated with lunar flight trajectories of various energies, azimuth angles, and injection angles with a breakdown of the critcalness of each error, and a corrective maneuver (4 dimension). 3. Analyze four dimensional flight profiles.	1. Improve techniques for establishing location in orbit or space.
LUNAR LANDING	1. Analyze ground support endeavors toward this mission.	
LUNAR LAUNCH AND TRANSEARTH FLIGHT	1. Analyze four dimensional flight profiles. 2. Analyze midcourse corrections to establish optimum time, place, and magnitude of correction parameters. 3. Analyze guidance accuracies associated with flight trajectories of various energies, azimuth angles, and injection angles with a breakdown of the criticalness of each error, and a corrective maneuver. (4 dimension).	
EARTH REENTRY AND LAND	1. Analyze four dimensional reentry flight profiles for reentry body. 2. Analyze responsibilities of ground support in reentry guidance.	

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[redacted]

(Continued)

TABLE 15. RECOMMENDED PROGRAM SUPPORT INVESTIGATIONS

ALL MISSIONS	1. Analyze manned vs. unmanned capabilities for each mission. 2. Establish abort guidance capabilities and limitations for each mission. 3. Analyze tradeoff considerations in all missions. 4. Establish estimated reliability. 5. Dual system analysis. 6. Analyze each mission for trajectory correction capabilities required. 7. Analyze guidance theoretical techniques to establish optimum mode.	1. Minaturization and microminaturization of systems. 2. Cryogenic systems development (gyroscope, accelerometers). 3. Improve sensitivity, accuracy and reliability of inertial systems. 4. Digitizing of initial systems. 5. Improve timing techniques.

[redacted]

GUIDANCE PROJECT G-1
HORIZON SENSOR IMPROVEMENT

1. Task Statement. To develop techniques to improve horizon sensor performance such that the horizon (or vertical) can be established to an accuracy of $\pm$ 0.1 degree or better on the dark side of a planet and moon.

2. Justification. Lunar and earth orbiting vehicles will require accurate attitude and alignment control in orbital launch, rendezvous, and docking missions. Current techniques do not maintain the desired accuracy on the dark side of the earth.

3. Present Status. Latest horizon scanners are accurate to $\pm 0.25^{\circ}$ although some are reported to be accurate to $\pm 0.1^{\circ}$ on the light side of earth.

4. Criticality. If possible this improvement should be made prior to the development of the guidance system to ensure compatibility.

5. Applicability. Important to earth and lunar orbital rendezvous, orbital docking and orbital launch missions.

6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-2

MICROMINIATURIZATION OF SYSTEMS

1. Task Statement. Microminiaturize or miniaturize as many subsystems of the launch vehicle guidance system as possible without affecting performance.

2. Justification. A decrease in size and weight of a launch vehicle system will result in substantial conservation of fuel for the lunar mission. Or, for a given allowable weight, consideration could be given to dual systems.

3. Present Status. Substantial progress is being made along these lines in electronic and inertial system design and development.

4. Criticality. Such a program is not essential to the successful accomplishment of the lunar mission and should not be undertaken unless there is substantial opportunity for success.

5. Applicability. All missions.

6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-3

DUAL SYSTEM ANALYSIS

1. Task Statement. Analyze the effect of dual systems, subsystems, and components in guidance system performances.

2. Justification. Guidance system reliability is not as great as it should be. It is usually considered to be the least reliable of the launch vehicle systems. Substantial effort has been placed on improvement of performance; however, improved reliability through dual system design may more than substantiate the resulting added payload. A complete system analysis would be required; tradeoff considerations would play a significant part in the analysis.

3. Present Status. As far as can be determined, none.

4. Criticality. If this project is to be undertaken, it should be considered early enough such that the results could be inserted into the lunar mission vehicle designs. This project could assure safe return of the manned lunar vehicle.

5. Applicability. All missions.

6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-4

CRYOGENIC SYSTEMS DEVELOPMENT

1. Task Statement. To develop cryogenic inertial systems to improve performance of launch vehicle guidance systems.

2. Justification. Research and preliminary developmental tests have indicated that there is promise of attaining the needed improvement in inertial system performance by the utilization of cryogenic techniques. Substantial effort is being placed on the development of such systems; it should continue.

3. Present Status. The research and development of cryogenic gyros, and accelerometers is being sponsored by NASA.

4. Criticality. It is doubtful that systems developed will be placed in the early lunar flight tests. A history of reliable performance will have to be attained first.

5. Applicability. If successful development will be applicable to second generation lunar and first generation interplanetary missions.

6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-5

DOCKING OPTICAL ALIGNMENT TECHNIQUES

1. Task Statement. To develop optical techniques for accurate alignment during orbital rendezvous and docking.

2. Justification. Alignment between vehicles of $\pm 0.1^{\circ}$ or better should be attained for spatial docking. Current attitude sensors have accuracies approaching $\pm 0.1^{\circ}$ — thus vehicle to vehicle misalignment of 0.2° could result. An optical or visual alignment technique involving direct vehicle to vehicle alignment measurement could improve this accuracy substantially.

3. Present Status. Some effort has been placed on this development and the results have been good. Very little additional effort is required to complete the R and D.

4. Criticality. The state-of-the-art alignment techniques for spatial docking are adequate; however, for very little effort substantial improvement could be made.

5. Applicability. Spatial docking.

6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-6

MIDCOURSE CORRECTION ANALYSIS

1. Task Statement. Analyze midcourse corrections for the various missions to establish the optimum and pessimistic correction magnitude time, frequency, place and philosophy for all possible flight profiles.

2. Justification. For every mission flight profile requiring correction, there is an optimum flight correction schedule. The criteria for optimum correction is that a minimum amount of propellant is used to successfully accomplish a mission. This optimum schedule of correction should be established and programmed into the flight profile of every mission in order to conserve vehicle fuel.

3. Present Status. Most studies heretofore have only established a need for midcourse correction. Some have analyzed specific correction philosophies, some have indicate probable magnitude of correction to be applied at arbitrary positions in space for specific missions. No attempt at continuity of study is evident.

4. Criticality. This study is an important prerequisite to launch vehicle design, especially for lunar missions in which conservation of fuel is so important.

5. Applicability. All missions involving corrective boost.

6. Reference. Analysis of Guidance

GUIDANCE PROJECT G-7

SPATIAL LOCATION TECHNIQUES

1. Task Statement. Develop new techniques or extend the state-of-the-art of current methods to establish spatial location accurately.

2. Justification. Current techniques for establishing position in space are not as accurate as required for most spatial maneuvers, especially those involving orbital transfer, rendezvous, docking, and lunar flights. Conservation of fuel is the primary reason for having accurate spatial location; proper correction maneuvers will result.

3. Present Status. Current techniques involving star and planet tracking are adequate for undertaking the manned lunar missions anticipated in the near future; spatial location estimates indicate current aaccuracies of ± 50 miles. Radar techniques, because of power requirements, black-outs, inherent topographical inaccuracies are not adequate.

4. Criticality. Current techniques are adequate to perform the first generation missions outlined.

5. Applicability. Spatical travel—translunar and transearth flights.

6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-8

FOUR DIMENSIONAL FLIGHT PROFILES

1. <u>Task Statement</u>. Conduct studies to establish four dimensional flight profiles for all missions. Time is the fourth dimension.

2. <u>Justification</u>. Most early moon-earth flight profile studies were two dimensional (planar); current studies are for the most part three dimensional. These latter studies have shown that the probable flight trajectories are not planar, have analyzed required accuracies to be able to undertake various lunar missions, and have established limiting flight parameters. Time of launch, time of flight, and coordination of time with other phenomena are other flight parameters affecting all missions whether in the earth to earth orbit scheme, earth to moon scheme, or moon to moon orbit scheme. These time parameters have not yet been ascertained.

3. <u>Present Status</u>. Little four dimensional flight profile effort has been undertaken.

4. <u>Criticality</u>. These profiles should be established as soon as possible. They will help establish launch vehicle computer design requirements and launch vehicle flight correction capabilities which affects propulsion system design.

5. <u>Applicability</u>. All missions.

6. <u>Reference</u>. Analysis of Guidance.

GUIDANCE PROJECT G-9

VISUAL DOCKING GUIDANCE TECHNIQUES

1. Task Statement. Develope television or other techniques for visual guidance during spatial docking.
2. Justification. Usual guidance techniques are not accurate over short ranges. Television will aid the crew of a manned craft in establishing short distances between mating vehicles accurately.
3. Present Status. Visual techniques of this type have been developed for numerous applications—should not present any problem.
4. Criticality. This development will only improve docking techniques; spatial docking will be able to take place without visual aid.
5. Applicability. Spatial docking; and spatial fuel, man and equipment transfer missions.
6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-10

COMMAND LINK GUIDANCE FOR DOCKING

1. Task Statement. Improve command link techniques for unmanned orbital docking guidance.

2. Justification. Early orbital docking flights will be unmanned—vehicle guidance will be handled by command link. For successful missions, the communication link performance will have to be improved. Greater accuracy, reliability, command rate, and variety of commands (expanded command language) will be required.

3. Present Status.

4. Criticality. This improvement will have to be made before docking mission flights are undertaken.

5. Applicability. Orbital docking missions.

6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-11

ALTIMETER IMPROVEMENT

1. Task Statement. To develope altimeter techniques or improve current performance such that altitude above the earth or moon can be established to 0.1% accuracy up to a range of 125,000 nautical miles.
2. Justification. Accurate range and range rate information will enhance successful missions in which position determination—altitude above the moon or earth is important.
3. Present Status. Doppler C.W. techniques under development supposedly have a 1% accuracy to 2000 nautical miles. Ionospheric backout occurs intermittently.
4. Criticality. Present capabilities are adequate for the mission indicated for Apollo and first generation vehicles. More sophisticated missions will require improvement.
5. Applicability. Earth reentry, orbital rendezvous, orbital docking, and lunar landing.
6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-12

MANNED VERSUS UNMANNED GUIDANCE

1. Task Statement. Compare the performances of manned and unmanned guidance for all lunar missions.

2. Justification. The added weight, cost, and probable lower reliability associated with automatic guidance techniques may not warrant its use on some or all of the manned lunar missions. (It may be better to utilize the weight for greater propellant loads). Automatic techniques will have to be developed for the unmanned flights—so development costs will still be evident. Perhaps automatic techniques should be used as a backup capability—or the manned system used as the back-up system.

3. Present Status. Some comparative studies of manned versus unmanned guidance for docking have indicated a significantly greater reliability associated with manned guidance.

4. Criticality. This is an extremely important guidance philosophy problem that should be resolved as early as possible in the manned lunar program. Unmanned Apollo missions will probably be instrumental in the decision.

5. Applicability. All manned missions.

6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-13

GUIDANCE AND CONTROL OF ORBITAL MATED ASSEMBLIES

1. Task Statement. Analyze the guidance and control system requirements of a mated assembly following docking.
2. Justification. The mating or docking of two or more vehicles, each having its own guidance system, into a single vehicle will necessitate an analysis as to the guidance and control system to be used, the requirements of each system before and after docking. Compatibility will have to be established.
3. Present Status. None.
4. Criticality. This should be done prior to the design of the docking vehicles. It could assure successful accomplishment of the docking mission.
5. Applicability. Docking missions.
6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-14

GROUND SUPPORT GUIDANCE FUNCTIONS

1. Task Statement. Analyze the ground support functions and requirements in each of the lunar missions.

2. Justification. Although ground support will supposedly be backup to the vehicle systems capability in the various lunar missions, the functions and requirements of ground support in these missions should be established in order that the ground support capabilities will be compatible with requirements.

3. Present Status. Opinions regarding the responsibility of ground support in the manned lunar missions vary from backup to vehicle systems—to important communication and command link functions.

4. Criticality. Decisions regarding ground support functions and requirements should be made as soon as possible to ensure that desired capabilities will be attained.

5. Applicability. All missions.

6. Reference. Analysis of Guidance.

[redacted]

GUIDANCE PROJECT G-15

RENDEZVOUS ERROR ANALYSIS

1. Task Statement. Conduct an error analysis associated with the rendezvous of vehicles in space.

2. Justification. The capabilities of each vehicle system involved in spatial rendezvous should be analyzed and the probability of having a successful mission should be established. Such an analysis will indicate systems requiring improvement in order to assure successful rendezvous.

3. Present Status. An error analysis of docking has been conducted— but this same technique should be carried out for rendezvous— without which docking cannot occur.

4. Criticality. Could assure a successful rendezvous mission.

5. Applicability. Orbital rendezvous mission.

6. Reference. Analysis of Guidance.

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GUIDANCE PROJECT G-16

SHORT RANGE-RANGE AND RANGE RATE DETERMINATION

1. <u>Task Statement</u>. Develop techniques and devices for the accurate measurement of range and range rate over short ranges (0-25 feet).

2. <u>Justification</u>. Current range and range rate techniques for short ranges are not as accurate as required for spatial docking, and earth and lunar landing. In landing, engine cutoff a number of inches off the earth or lunar surface could cause substantial damage to the vehicle in the subsequent fall. Special docking operations may require monitoring the distance between vehicles accurately over the last few feet of separations prior to mating.

3. <u>Present Status</u>. Current techniques do not cover the range 0 to 4 feet. Visual and photocell techniques show promise but have not yet proven to be acceptable. If moon is covered with dust, blurred vision, caused by the dust may make these techniques unusable.

4. <u>Criticality</u>. This development may not be required for spatial docking. Such a device should definitely be developed prior to the lunar landing mission.

5. <u>Applicability</u>. Soft lunar landing, and spatial docking missions.

6. <u>Reference</u>. Analysis of Guidance.

GUIDANCE PROJECT G-17

ABORT GUIDANCE

1. Task Statement. Establish guidance limits for which abort will occur. Analyze the guidance of the vehicle following mission abort.

2. Justification. Safe return of the vehicle crew is mandatory. Abort of a mission can occur for numerous reasons and there will have to be limiting conditions under which abort is mandatory, conditions under which it is probable, possible, etc. An analysis of abort during all missions will establish the abort sequence and define abort system requirements.

3. Present Status. Very little effort has been accomplished in the analysis of abort for lunar missions.

4. Criticality. This is a definite prerequisite to the design of abort systems for the lunar missions.

5. Applicability. All lunar missions.

6. Reference. Analysis of Guidance.

GUIDANCE PROJECT G-18

GUIDANCE RELIABILITY ANALYSIS

1. Task Statement. Establish the anticipated and required guidance system reliability for the various lunar missions.

2. Justification. Guidance systems are considered the least reliable system of the launch vehicle systems. To attain a high level of mission success, the anticipated system reliability should be compatible with required reliability. This analysis will pinpoint system components and subsystems that should be improved to attain the desired level of performance. Dual system and inter-system trade-offs would be established.

3. Present Status. Limited effort has been undertaken in this area.

4. Criticality. This study should be conducted early in the program (prior to design of the guidance system) and should continue through the lunar program so that latest pertinent information would be included in the analysis.

5. Applicability. All missions.

6. Reference. Analysis of Guidance.

		Funding	Contr. (In-house Lab.)	
1.	Cryogenic Gyro[illegible]	[illegible]	[illegible]	[illegible]
2.	Development of Light Pickoff Using Lateral Photocell Effect	FY61/$ 35K	GM&C Div.	[illegible]
3.	Hydrodynamic Gyro[illegible] Development	[illegible]	[illegible]	[illegible]
4.	Air Bearing [illegible] Develop[illegible] Machining Techniques and Applications.	FY62/[illegible]	[illegible]	[illegible]
5.	In-house development of High Accuracy Gyroscopes	[illegible]	[illegible]	Applic[illegible]
6.	New Air [illegible] Cylin[illegible] Gyro[illegible]	FY62/$ 50[illegible]	GM&C	Applicable [illegible]
7.	[illegible] and [illegible]	FY62/[illegible]	[illegible]	[illegible]

[redacted]

CATEGORY: GYROSCOPE RESEARCH AND DEVELOPMENT (Cont'd)

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
8.	Cryogenic Accelerometer in Inertial Systems	FY62/$150K	G&C-R	Applicable. Undertake.
9.	Miniature Air Pump Development	FY62/$ 40K	M-G&C-R	Applicable. This seems to be a purchase request for an item already developed.

General Remarks: As indicated in the Analysis of Guidance, the state-of-the-art of gyroscopes has not attained that required to undertake the various lunar missions without relying upon midcourse correction capabilities. R & D effort to improve this state-of-the-art/requirement incompatibility should be continued. However, it should be pointed out that these new guidance systems will probably not be ready to be incorporated into launch vehicle guidance system designs for the early lunar flights.

[redacted]

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF GUIDANCE

CATEGORY: GUIDANCE STUDIES

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Ascent Guidance Studies	FY62/$200K	Adv. St. Brch. G&C Div.	Definitely applicable to manned lunar mission. A similar study should be conducted over the other phases of the manned lunar mission, i.e., lunar landing, lunar orbit, etc.
2.	Orbital Launch Guidance Studies	FY62/$ 50K	Adv. St. Brch. G&C Div.	Applicable to interplanetary flight only—it is <u>not</u> applicable to the manned <u>lunar</u> mission. This study is a prerequisite to manned interplanetary missions.
3.	Reentry and Return Guidance Studies	FY62/$100K	Adv. St. Brch. G&C Div.	Definitely required and applicable. Previous studies have only touched on this area.
4.	Orbital Launch Guidance System Studies	FY62/$150K	Adv. St. Brch. G&C Div.	Definitely required and applicable. This should be extended to include lunar launch missions.

<u>General Remarks</u>: Close coordination of the projects discussed above with the Apollo program should be undertaken to avoid duplication of effort.

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF GUIDANCE

CATEGORY: GUIDANCE AUXILIARY EQUIPMENT RESEARCH AND DEVELOPMENT

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Special Timing Networks	FY61/$25K	G&C	Applicable. More information is required to ascertain if an adequate device already exists.
2.	Peltier Cooling of Infrared Detectors and the Development of Semi-conductor Heating and Cooling Systems	FY61/$64K	G&C	Applicable. Undertake.
3.	Peltier Device	FY62/$50K	M-G&C-R	Applicable. Undertake.
4.	Development of New and Improved Techniques for Use in Horizon Sensing Devices	FY62/$30K	M-G&C-R	Applicable. Definitely undertake.
5.	Advanced Studies on the Three Axes Motion Simulator (Equipment Required for In-House Efforts)	FY62/$30K	M-G&C-R	Applicable.

<u>General Remarks</u>: None.

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF GUIDANCE

CATEGORY: GUIDANCE ELECTRONICS RESEARCH AND DEVELOPMENT

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Digital Input/Output Devices	FY61/$100K FY62/$100K	G&C	Applicable, but what devices—accelerometers, rate gyros? Or perhaps just digital converters with special applications? There are definite advantages to digitizing guidance devices, but there is not sufficient information presented to determine if this will result in duplication.
2.	Miniaturization and Reliability of Missile-borne Heavy Duty Contractors	FY61/$ 39K	G&C	Applicable, undertake.
3.	Electronic Circuitry Micro-miniaturization Development	FY61/$ 80K	G&C	Applicable. This project seems to be a general state-of-the-art improvement subsidy which could be undertaken under any program. It will have many applications. Undertake.
4.	Development of Semiconductor Devices	FY61/$ 80K	G&C	Applicable. Again, this project seems to be a state-of-the-art improvement subsidy with many applications. Undertake.
5.	Microminiaturization of Circuits and Development of Special Semi-conductor and Magnetic Devices	FY62/$400K	G&C Div. M-G&C-R	Applicable. Undertake. Carries on effort of #3 and #4.

CATEGORY: GUIDANCE ELECTRONICS RESEARCH AND DEVELOPMENT (Cont'd)

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
6.	Digital Filters	FY62/$400K FY63/$100K	G&C Div. Nav. Branch	Applicable. Undertake

<u>General Remarks</u>: Generalized effort, such as that apparently undertaken in #3, should be monitored closely. Lack of specific goals (state-of-the-art improvements) or specific applications can result in little progress (value per dollar).

V. CONTROL

INTRODUCTION

5.1 The technical area of control is the amalgamate of all the control systems; the systems which translate and impart the necessary electrical or mechanical energy to the propulsion components or control surfaces to provide the desired flight path, alignment, or velocity in space. The major systems involved in the control functions are:

a. Hydraulic gimbal system

b. Reaction attitude control system

c. Vernier velocity control system.

5.2 The command input to the control systems will originate from the analog control computer which contains the servo-electronics for positioning the control elements (engine-throw angles, attitude-control jets, etc.) in pitch, yaw, and roll in accordance with the guidance system commands and the error signals resolved from the platform gimbal angles. The control computer also contains the electrical shaping networks required for rigid and flexural stability constraints.

5.3 Guidance commands to the control computer will probably be in analog form and previously sequenced by the guidance system. The flight control computer will perform:

a. Monitoring of thrust vector direction and magnitude.

b. Translation of any guidance command into gimbal position or valve flow.

c. Comparison of the guidance requirement with the position of the component at any instant.

d. Amplification of any difference involved between requirement and current status.

e. Superimposition of any constraints or damping characteristics upon any amplified difference.

f. Choose the proper component or combination of components to give the desired attitude result upon command.

g. Command the pertinent components involved to act upon the vehicle according to the magnitude.

The requirements and capabilities in the flight control computer will be discussed at length within the technical area of data processing.

5.4 Control system outputs are essentially the application of thrust vector control techniques; the recipients of this action will be components such as the main engine gimbal hydraulics, vernier engine control valves, or the attitude adjustment jet solenoids. The actuation of these control components invariably results in a change in attitude, velocity, or trajectory of the vehicle.

5.5 In terms of the manned lunar mission, the boundaries of control modes and techniques are well established. The control system must be available to implement the commands of the guidance system through the basic control patterns. These commands will involve:

a. Flight disturbance control.

b. Mission flight plan control.

The control systems must be able to handle the summation of the requirements in these two sequences since they do not encompass exclusive time frames. The difference between these two sequences is discussed in subsequent paragraphs.

5.6 Flight disturbances will be present to some degree in all mission sequences. To simplify the analysis of requirements, the disturbance variables will be analyzed as a single sequence and the resultant requirements will be superimposed upon the flight sequence (steering) data in the final prediction of the control system technical area plan.

5.7 If a vehicle could fly an undisturbed flight path, the major components of the control system would be relatively idle; the flight path control uses only a relatively small percentage of the design capacities of the control components. A large reserve capacity is designed into control to cope with the flight variables that are either unpredictable, or unreliably so.

5.8 It is true that disturbance control, while it could be called a secondary function to steering, dictates most of the control system magnitudes of capability, such as reaction time and gimbal angle throw requirements. It also seems that high accuracy would not be a major requirement since the nature of flight disturbance control would be one of semi-emergency nature; the urgent job is to smooth the disturbance immediately and take time to make adjustments later.

5.9 On the other hand, mission flight plan control is concerned with accuracy and the ability to attain reliable control of a vector angle or magnitude within severe limits. The length of the flight path involved and the restrictions on fuel available for midcourse and terminal corrections will provide stringent inputs to the accuracy requirements of the control systems.

5.10 The missions referred to in mission flight plan control are the separate sequences listed below that represent step functions or secondary missions in attaining the primary mission of the manned lunar expedition.

Manned Lunar Missions

a. Earth launch and orbit.

b. Orbital rendezvous.

c. Orbital docking.

d. Orbital assembly, maintenance, and checkout.

e. Earth orbital launch and translunar flight.

f. Lunar orbit and landing.

g. Lunar launch and transearth flight.

h. Earth re-entry and land.

Judicious choices between the content, combination, and sequence of the above missions will provide all the possible flight tests, probes, and manned lunar attempts within the manned lunar program.

5.11 The mission referred to above as orbital assembly, maintenance, and checkout (d) will not be included in the analysis of control problems as it is not considered applicable within the technical area of control systems.

5.12 The same control techniques and control problems are utilized in several of these missions. For example, the attitude and velocity control problems are similar in any midcourse correction control sequence whether the trajectory is outbound, inbound or orbital transfer. The magnitude, time, and accuracy requirements may be somewhat different, but the general problems and techniques remain the same, independent of the chosen missions. For this reason, the control requirements will be discussed in terms of sequences or problem groupings instead of the missions. These consolidated missions or areas of similar control problems are listed below.

a. Earth launch and orbit.

b. Rendezvous.

c. Docking.

d. Orbital orientation and launching.

e. Midcourse correction.

f. Lunar maneuvers.

g. Earth re-entry and landing.

Table 16 shows the relation between the control problem areas (which are the basis for discussion in this section) to the Manned Lunar Missions (which are the common denominators of the report as a whole).

5.13 The three major control systems have fairly specific areas of application within the control problem areas, but they should not be restricted in this sense. Later discussions will indicate the probable use of the control systems involved in the various mission control functions.

5.14 In order to establish the system requirements within the technical area of control, there must first be a listing and examination of the sequences or events in which control of the vehicle is critical to the success of the mission. Listed in paragraph 5.15 are the events in which control is a critical function. Each area will be investigated and discussed in terms of expected level of disturbance and control requirements for all control systems. This analysis will generate the technical-area plan.

TABLE 16. CONTROL AREA/MISSION RELATIONSHIPS

CONTROL PROBLEM AREA	MISSION
1. Midcourse Correction includes:	1. Orbital Rendezvous 2. Earth Orbital Launch and Translunar Flight 3. Lunar Launch and Transearth Flight
2. Lunar Maneuver includes:	1. Lunar Orbit and Landings 2. Lunar Launch and Transearth Flight
3. Earth Launch and Orbit include:	1. Earth Launch and Orbit
4. Rendezvous includes:	1. Earth Orbital Rendezvous
5. Orbital Orientation and Launching include:	1. Earth Orbital Rendezvous 2. Earth Orbital Launch and Translunar Flight 3. Lunar Launch and Transearth Flight
6. Docking includes:	1. Orbital Docking
7. Earth Re-entry and Landing include:	1. Earth Re-entry and Landing

5.15 The following discussions of control problems will be in two parts:

Part I. Mission Flight Path Control

A. Earth Surface Launch and Orbit

B. Rendezvous

1. Outbound
2. Inbound

C. Docking

D. Orbital Alignment and Launch

1. With Platform
2. Without Platform

E. Course Corrections

1. Midcourse
2. Terminal

F. Lunar Maneuvers

1. Flyby
2. Orbit
3. Landing
    a. From Orbit
    b. Direct
4. Lunar Launch
    a. To Orbit
    b. Direct Escape

G. Earth Re-entry

1. Direct
2. Orbital

Part II. Flight Disturbance Control.

A. Engine Out

B. Control Actuator Malfunction

C. Winds

D. Staging

E. Bending

F. Sloshing

A third part will consist of a system development outlook or achievement discussion in relation to the major control systems in the launch vehicle:

Part III. System Developmental Outlooks.

PART I. MISSION FLIGHT PLAN CONTROL

5.16 When the various disturbances are removed from a vehicle flight program, the control systems are left with the responsibility of implementing the desired characteristics of the vehicle in attitude and velocity.

5.17 The analysis of the requirements of these control systems will be made by examining the various control problem areas to find the predicted flight parameters and conditions that the control systems must adhere to.

5.18 The three major control systems will all be used at one time or another during the lunar mission. A short description of the systems will assist in reviewing the control analyses.

Mainstage Hydraulic System

5.19 The thrust vector control for the main engines cluster; used almost exclusively in the first two stages of the earth booster and for the orbital launching stage. The vernier engines are similarly hydraulically controlled but will be included as another system since the vernier engines are used in many more sequences than the mainstage engines and their associated hydraulic controls. This is primarily due to limited restartability of the mainstage engines.

Reaction Jet Systems

5.20 The small, pulse regulated, hypergolic bipropellant chambers used for attitude control or extremely small linear adjustments in free fall space. There will be choices between high and low thrust attitude jet systems for coarse and fine altitude adjustments. Initial alignment of the mainstage and vernier engine thrust vector just before ignition will be by the reaction jet systems. The reaction jets will probably not be used extensively during thrust periods when gimbal control is available.

Vernier Velocity Control Engines

5.21 The vernier velocity engines are gimbaled medium level thrust engines using hypergolic bipropellants. They will be used for tasks such as velocity midcourse corrections, lunar orbital injection, and providing thrust vector control along with part of the lunar launching impulse. The engines probably will be hydraulically controlled but possibility exists for pneumatic or mechanical control. The mainstage engines will probably be unable to provide restartability so the need for the vernier control engines is evident.

5.22 Table 17 shows the probability for use of the various control systems throughout the lunar flight. This information is partly logical and partly derived from various vehicle configurations and designs available as bids or statements of task concerning the Apollo program.

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Table 17. Probability of Use

CONTROL PROBLEM AREA	CONTROL SYSTEM		
	Mainstage Hydraulic Gimbal	Reaction Jet	Gimbaled Vernier
Earth Launch and Orbit	Hi	Low	Possible
Rendezvous	Possible	Hi	Hi
Docking	Lo	Hi	Lo
Orbital-Launch-Alignment	Lo	Hi	Lo
Midcourse Correction	Possible	Hi	Hi
Lunar Maneuver	Possible	Hi	Hi
Earth Re-entry and Land	Lo	Hi	Lo

NOTE: The "possible" probabilities indicate that the system is capable of use in that function and/or provide back-up or some rought thrust vector control with another system providing the more accurate adjustments.

Earth Surface Launch and Orbit

5.23 During earth launch the flight plan is relatively simple; there is approximately a 10 second vertical rise before the vehicle is programmed into a small angle of attack. Roll control is usually established in this period to orient the vehicle geographically before inducing the small angle of attack. Staging occurs at approximately 200-250,000 ft; the vehicle second stage burns into the 100 nautical mile orbit with possible angles of attack up to 17° in the orbit injection cutoff.[2/]

5.24 Engine gimballing is the control method in the flight plans in the lower and upper atmosphere. The roll control can be carried out by the gimbaled main stage engines or by tangentially mounted reaction roll rockets on the booster. Addition of roll rockets on the S-I stage would result in added lift off weight and would be an unnecessary solution since roll control gimbaling capacity is available anyway. The roll control system on the S-II stage could be used if a control jet system is readily available on the exterior of the stage. The vehicle and the space craft payload will orbit until initiation of Hohmann transfer into 300 mile orbit.

2/ Part I. NASA Industry Apollo Technical Conference, July, 1961, CONFIDENTIAL.

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5.25 The additional velocity required for orbital transfer will be provided by vernier engines, capable of providing the approximate 700 fps needed, or possibly by restart of the main propulsion engine using fuel left over from the orbit injection flight. In either case, if the engine is liquid fueled, the problem of ullage control under zero gravity will be present. Since the fuel tanks are not full, the fuel and vapor do not hesitate to mix and float about the tank while in weightless condition.

5.26 Successful ignition can be prevented by vapor pockets in the turbopump and injection ports. Initiation of thrust at approximately .1 to .05 g[3]

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[3] Lockheed Georgia Co., Report ER 5388, October, 1961, CONFIDENTIAL.

is sufficient to orient the fluids properly for successful engine ignition. Time required to settle the liquids is variable and depends roughly on the g forces involved and degree of interspersion of vapor and liquid. This low thrust tends to perturbate the initial orbital launch conditions and must be considered in the launch sequence. Ullage availability of 0.1 g would result in the perturbations prior to main ignition shown in Table 18. It could be supplied by rearward attitude type nozzles if they were capable of the thrust level, or by solid propellant packages triggered in the ignition sequence.

5.27 Longitudinal ullage control provides the simplest plumbing requirements in the fuel feeding lines, the propellant being drained straight out the end of the tank opposite the reaction. But long term longitudinal thrust tends to disturb the initial launch orbit.

TABLE 18

Ten Second Longitudinal Ullage Control Thrust
Perturbation Prior to Orbital Launch

Acceleration Rate	Resultant Δ Velocity	Resultant Distance
0.1 g	32.2 ft/sec	161 ft

5.28 Induced spin will orient the propellants without orbital disturbance, but the fluid would have to be drawn from the side of the tank complicating the plumbing problem by at least one 90° bend in the fuel feed lines. The rate of spin would be low but would effectively neutralize the attitude control jets mounted on the periphery unless rotational switch gear was incorporated into the flight control computer. There are some other ways to separate the fluids from vapor but these are basically mechanical diaphrams which are difficult to maintain or replace in the sealed tanks. Expulsion bags or diaphrams incur reliability and weight penalties but are a possible solution for non-thrusting ullage control. Solid rocket systems are certainly feasible and have been developed to the output levels necessary for ullage control.

Orbital Rendezvous

5.29 Since the second stage will have approximately half of the space craft-orbital launch vehicle assembly as a payload, the Hohmann transfer will have to terminate under conditions that allow compatible rendezvous.

The attitude control system should align the stage before ignition so that the main applied thrust vector will operate tangentially to the 100 nautical mile orbit path after ignition. Allowable error in this alignment should be capable of not requiring a midtransfer correction of more than 200 ft/sec. 4/ There is nothing gained in having the orbiting stage continually tangentially oriented in orbit before transfer. Plenty of time is available to orient just before ignition at a substantial saving in fuel.

5.30 Attitude control can be implemented continuously at the onset of vehicle to vehicle tracking for orbital transfer while still in the 100 nautical mile parking orbit, or it can be implemented only before the events requiring attitude orientation and alignment (such as periods of thrust).

5.31 Minimum attitude control would enact 6 corrections (3 thrust periods during transfer and 3 thrust periods prior to docking maneuver). The usual correction for each event would be around 90°. If the vehicle were continuously controlled at onset of tracking to within 1° of tangential attitude, about 4 corrections per minute would be required for the 60-70 minute transfer. Power and fuel savings are quite evident in the comparison. Table 19 documents some of the system variables for the two control sequences. This preliminary study is not a basis for judgment, but an attempt to show the nature of the guideline studies which are important inputs to the engineering-hardware solutions.

TABLE 19

Attitude Control Requirements for Transfer and Rendezvous Prior to Docking

Control Mode	Maximum Correction Required (s) degrees	Corrections Required	Maximum Time Between Corrections	Acceleration Capacity Required
Continuous Control	1°	~270	15 sec	Relatively High
Minimum Control (Incremental)	$\sim 90^\circ$	6	20 min	Relatively Low

4/ General Dynamics/Astronautics, Report No. AE 61-0967, October, 1961, [redacted]

[redacted]

[illegible]

[illegible] time [illegible]

[illegible] detection [illegible]

[illegible]

[illegible] 2 [illegible]ows.

[illegible]

[illegible]	[illegible] Pulling [illegible] (lbs)	[illegible] Transfer [illegible]	[illegible] Weight (lb[illegible]	[illegible]	[illegible]
[illegible]BOUND	+25,500	+[illegible]	334,[illegible]	[illegible]	
[illegible]OUND	-12,000	-700	20,0[illegible]	[illegible]	

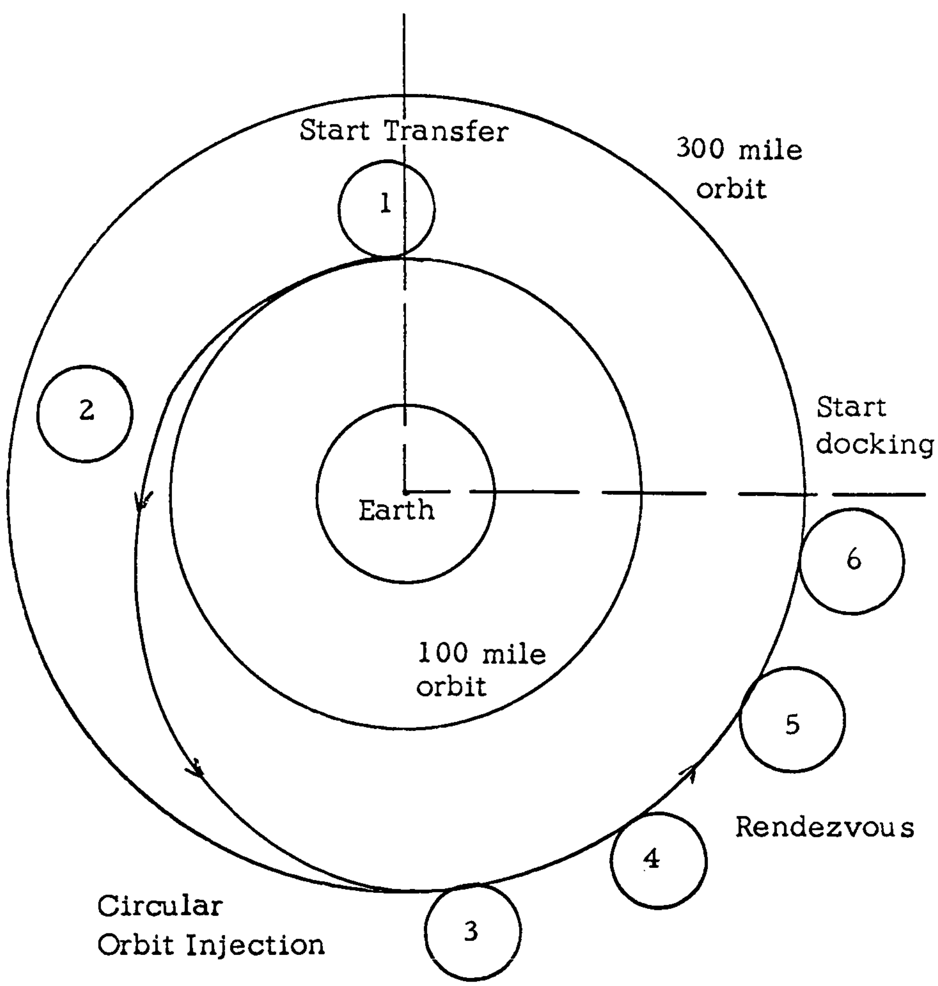
5.35 Other problems besides [illegible] capability [illegible] consid[illegible] nebulous enough to prohibit inbound [illegible] or [illegible] time. In terms of control components, however, [illegible] be sa[illegible] once acceptable [illegible] is [illegible], [illegible] existi[illegible] systems are easily [illegible] to the inbound [illegible] and [illegible] quence.

[redacted]

TABLE 20

Major Attitude Control Function During
Orbit Transfer

Pulse No.	Reason for Correction	Predicted Correction
1.	Initial transfer thrust	?
2.	Mid transfer correction thrust	90°
3.	Terminal circularizing thrust	90°
4. 5.	Rendezvous correction thrust	30°
6.	Initiation of docking control thrust	30°
		Total 270° + ?



Docking

5.36 In order to impact with a minimum of momentum between the two vehicles, the attitude control system will have to operate with minimum thrust to accomplish the motion required within the last few feet of travel. This requirement calls for a variable thrust control system with a minimum of 2 relative levels of thrust.

5.37 At the present time, the Mercury orbital vehicle uses two separate systems, one high-one low. This is one of the most direct solutions and provides a back-up system.

5.38 As attitude jet system reliability improves, the combination redundant-low thrust system could possibly be replaced with a single variable thrust system.

5.39 Wide range variable thrust is not particularly necessary and would probably complicate the attitude jet system unnecessarily. Analysis of the vehicle moments about the 3 major axes combined with control rate requirements should evolve several optimum incremental stages of attitude control thrust to be used in various sequences of the flight-rendezvous docking mission.

5.40 If the sensors and the guidance closed loop systems operate at the design levels, the control system will be capable of providing the required reactions without significant changes from current operating techniques.

5.41 Since control reaction would probably be almost continuous at docking, system effects including heat transfer from long periods of use and power requirements when all nozzles are in operation should be studied for inclusion in developmental designs.

5.42 Inbound rendezvous and docking is a distinct possibility in time, but seems unlikely in first generation lunar return and re-entry missions. The problems involved will be quite similar but success of the inbound maneuver will be limited by the fuel requirements and tracking requirements necessary for injection into the proper parking orbit plane and altitude for inbound rendezvous with the orbiting platform.

5.43 The altitude difference is critical. As the difference in altitude decreases, the required orbital "parking time" increases for correction of phase differences between target and chaser; as the differential altitude increases, the Van Allen radiation hazard becomes more intense and unfavorable for human survival above 300 nm orbit altitude. There would be no significant radiation in an inbound parking orbit below 300 miles. The minimum altitude possible for a parking orbit is about 90 miles.

Orbital Launch Alignment

5.44 The space vehicle will be launched on translunar flight much the same as for transfer from parking orbit to rendezvous orbit. The orbital launch vehicle may be assembled from at least two payloads mated in the rendezvous orbit (Saturn) or may be in a direct flight vehicle (NOVA). The accuracy of the orbital assembly and checkout techniques are instrumental in establishing the control requirements for the periods of thrust as the vehicle is injected into translunar flight. Normal error can provide disturbing torques and introduce flight path perturbations that require control adjustments, which themselves can introduce more disturbance if the misalignments are severe enough.

5.45 The alignment sequences just prior to orbital launch will be identical with those in the parking orbit. Low g ullage control will be needed if the fuel tanks are not diaphramed to orient the fuel under zero gravity. Similar orbital launch can occur from lunar orbit towards Earth. The control problems are similar to those already discussed.

5.46 There is evidence that the Earth orbital launch of second generation vehicles toward the moon might be made with the assistance of a manned or unmanned launch platform. This could assist appreciably in initial reference alignment but would not provide any assistance to ullage control or attitude control after launch. It is doubtful that a platform would attenuate any control requirements for orbital launching.

Midcourse and Terminal Corrections

5.47 Course correction has been discussed in the rendezvous function where the correction takes place during the orbital transfer phase. The elements of the discussion are much the same for the transfer trajectory between the earth and the moon. The translunar path involves higher velocity, longer trip time, and greater distance. As would be expected, the control requirements increase, but not overwhelmingly. Where there was time for one midcourse correction in the orbital transfer, there is now time for three or four or more if adequate fuel is available. The performance of the orbital launch is of prime importance in determining the control requirements on the translunar and transearth flights. This particular problem area requires extensive systems analysis work, not just trajectory studies and error source discussions, but attempts to establish meaningful and logical models with which to ascertain the tradeoff effects of guidance, control, propulsion, and tracking subsystems.

5.48 In the translunar and transearth trajectories it might be proven that a particular level of component error is not objectionable if the weight saved by allowing the error to remain would permit the addition of enough midcourse correction ability to correct the results of the original error and have ability left to correct the presence of additional errors from other systems or components. The net result being, that the mission outcome has possibly been improved by the toleration of error within the system as a whole.

5.49 The major form of midcourse correction will be velocity addition or subtraction at some point in the actual trajectory so that the points of impact of the actual and desired trajectories will coincide even though the paths do not.

5.50 The time frame involved in applying ΔV correction is important in the same manner as in the discussion of attitude control requirements in the section pertaining to the orbital rendezvous maneuver.

5.51 Most all possible flight missions (lunar flyby, lunar orbit, and lunar landing) require similar techniques for midcourse corrections in flight. Accuracy requirements would differ with the most rigid probably being a direct lunar landing and the least restrictive being lunar flyby.

5.52 In examining the velocity characteristics of a passive body between the earth and the moon, it can be seen that the velocity falls remarkably due to gravity pull from the earth on the way to the moon and would rise in the same manner on the way back.

5.53 The lowest velocity in both flights is at the gravitational "midpoint" between the moon and earth. The velocity at this point would, theoretically, be the residual velocity over escape velocity that the vehicle left the earth or moon with at launch. This low velocity point is the point at which steering correction would be made most economically. The higher kinetic energies that resist change in the flight path direction are on each side of this point.

5.54 At this stage, it is possible to note how control system requirements for midcourse correction can be vitally dependent on many things, such as:

a. Perturbations resulting from orbital launch.

b. Allowable time frame for ΔV correction.

c. Predicted magnitude for ΔV correction.

d. Number of corrections planned.

e. Degree of flight path angle change attempted at or near gravitational midpoint.

f. Mission requirements near target.

It would not be realistic to predict requirements without some test data inputs as rough guidelines in the above considerations. However, on the basis of logical engineering parameters, it could be possible to predict some relative changes in requirements in relation to the above areas of consideration. Table 22 relates some of the trade-offs for the midcourse correction techniques or systems and their proportionalities to the flight variables shown.

5.55 While midcourse corrections will place the space craft on a path to a point in space, terminal corrections will be required to provide any orbiting impulses or enhance the successful return to earth on a flyby mission. The largest terminal correction will be from translunar trajectory into orbit about the moon. The requirements would remain independent, to some degree, of initial conditions since an orbit can be established from many initial conditions of velocity and altitude with roughly the same retro impulse. Prior errors before terminal correction would manifest themselves in orbital altitude and plane angle error. Orbital altitude requirements would have a wide range for a lunar orbit mission and a narrow range for a lunar landing mission. Since the orbit mission would logically be attempted before the landing mission, some concrete data input would provide strong guidelines to the maximum altitude error which can be expected prior to lunar orbit injection.

5.56 Orbital plane angle would be less critical for a landing mission and more critical for a flyby or lunar orbit mission since the error in plane angle could be corrected by lunar launch; but is not as easily corrected while the vehicle is orbiting about the moon on an uninterrupted flight. On return to the earth, terminal correction would be for orbiting, rendezvous or direct re-entry.

5.57 The primary problem area in early flight tests will be to find the trade-off level for the terminal correction capacity in relation to the probabilities of expected error at arrival.

Lunar Maneuvers

5.58 The terminal course correction in the vicinity of the moon will result in one of three conclusions to the earth-moon trajectory: (1) small corrections enhancing the circumlunar or flyby trajectory, (2) a circularizing impulse resulting in a lunar orbit, and (3) a direct entry with retro to a soft lunar landing. These maneuvers will be discussed below in terms of control requirements.

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TABLE 22
VERNIER VELOCITY MIDCOURSE CORRECTION SYSTEM REQUIREMENTS
FOR SOME FLIGHT VARIABLES

VARIABLES		DEPENDENT REQUIREMENTS: REQUIRED THRUST	WEIGHT: Fuel	WEIGHT: System	VECTOR ALIGNMENT ACCURACY	REQUIRED RESTARTS
1. Perturbances from Orbital Launch	HI	Higher	Higher	Higher		
	LO	Lower				
2. Allowable Time Frame for ΔV Correction	LONG	Lower		Lower		
	SHORT	Higher		Higher		
3. Predicted ΔV Correction	HI	Higher	Higher	Higher	Higher	
	LOW	Lower	Lower	Lower	Lower	
4. Number of Corrections Planned	ONE	Higher		Higher	Higher	Some
	MORE	Lower		Lower	Lower	More
5. Exploitation at Gravitational Midpoint	NONE	Higher	Lower			Some
	SOME	Lower	Higher			More
6. Mission Requirements	FLYBY		Lower			Some
	ORBIT		Higher			More

~~CONFIDENTIAL~~

5.59 The lunar flyby mission is probably more rigorous than is generally believed. The midcourse correction capabilities are limited, but must be able to correct the small trajectory errors which originate at orbital launch from earth and deflect the elliptical flight plan until it returns to Earth again. This is a much longer error effect time than any other lunar mission possesses.

5.60 As was pointed out in the discussion on Astronautics, any extremely small errors in launch can produce almost magnitude errors in pericynthion. A high pericynthion in itself is not hazardous, but because elapsed flight time determines the point of return (if any) to the Earth's surface, this deviation can be very critical.

5.61 If the initial early trajectory of the flyby mission can be held to high accuracy, or if the mission can be aborted if the total error is marginal for success, then the velocity terminal adjustment to the trajectory will be compatible with the hardware available to the mission.

5.62 Since there is not a whole lot of difference in the control actions for flyby and for lunar orbit,[5/] the same statement is true for lunar orbiting maneuvers. The lunar orbit injection would be characterized by a 3000 fps retro maneuver for lunar capture. This retro maneuver would require attitude orientation prior to thrust and thrust vector control during the retro period. These are capabilities which should pose no technological problems that are predictable at this time.

5.63 Generally, as the mission requirements move into the area of the lunar maneuvers, the major control problems shift from areas such as stability, reaction time and system capacities to the areas of environmental effects on system components, reliability, and elapsed time effects on the system performances.

5.64 The problem of hydraulic system fluid sensitivity becomes apparent even before earth orbital launch. As mission time elapses total effects of radiation on organic piston seals, diaphrams, and flexible insulation could become noticeable. Meteoroid erosion will not be serious unless enough energy is present to puncture a tank, pneumatic components, or solid rocket cases. These probabilities are low.[6/]

5.65 Space vacuum environment is particularly dangerous to the engine gimbal hinges and their lubrication. The coefficient of friction is a critical value in designing gain rates and energy requirements for gimbaling under

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5/ Same principles involved, only thrust and vector control magnitudes differ.

6/ Whipples distribution for Sporadic Meteoroids.

thrust. There are no values for the friction values under thrust in the space environment for such gimbal systems.

5.66 Temperature extremes and cycling is very damaging to pneumatic seals, hydraulic O-rings, and other similar materials. Recent study[7] shows that typical high temperature pneumatic seals will not fail[8] at temperatures to 1000°F but that they fail upon cooling to 650° to 700°F. Hydraulic seals are sensitive in the same way but will last up to several times as long as the pneumatic seals. There is also sensitive interaction between the synthetic hydraulic fluid compounded to withstand high temperatures and the deterioration of such fluids on the O-rings and seals. These effects are negligible on the short-life launch vehicles, but will become critical problem areas on the space vehicle due to multiple engine starts,[9] longer environmental effect times, and the need for continued high system reliabilities as mission time elapses.

5.67 In summarizing the responsibilities and problem areas for lunar fly-by or lunar orbiting, it seems that accuracy of vector control and timing are important in the early translunar injections and that the terminal lunar maneuvers are of similar nature, their difficulty being directly proportional to the success of the aforementioned translunar injection. The area of system-environment-reliability interactions is the major problem area for control components in lunar terminal maneuvers.

Lunar Landing

5.68 The early manned landings on the moon will probably be from a lunar circular orbit established at about 100 miles altitude. The advantages of the orbital landing would be opportunities to choose more precisely the general area of letdown with more certainty than would be possible under direct entry. In terms of the control systems, the two methods are similar enough to say that the capability for one should provide near capability for the other, and vice versa. The last 75 to 100 miles is the most critical in the lunar landing mission whether it be from orbit or direct entry.

5.69 Since other areas of endeavor will probably show an orbital landing to be more feasible for early flights, this discussion will bear primarily on that problem.

5.70 After injection into the desired orbit, a transfer thrust to an elliptic is affected which will impact the space craft on the surface of the moon. The

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[7] WADC Tech. Report 59-428, William Walker, April, 1961.

[8] Under N_2 300 psig.

[9] For midcourse corrections.

landing is softened by retro thrust resulting in velocity loss of around 600 feet/sec.[10] The optimum retro maneuver provides simultaneous arrival at zero altitude with zero velocity. Actually, the zero velocity point will probably be about 100 feet off the surface and the lateral residual velocity should be 20-30 fps. The last 100 feet will be under the control of a terminal guidance technique which is reasonably immune to the basic problems of low altitude such as rocket flame or dust attenuation of altitude sensors accuracy within 100 ft range.

5.71 The problem of controlling this downward deceleration from 100 miles to 100 feet has been examined and one major conclusion is presently critical.[11] A constant thrust retro engine is of marginal value due to difficulty in reliability of the basic engine parameters such as specific impulse and thrust. Such parameters as altitude and weight of the vehicle at lunar arrival are subject to uncertainty which yield considerable difficulty in trying to program a safe descent with a constant thrust system. Total expected error from a constant thrust rocket motor would be about 3.3% in velocity and 6.8% in altitude for a Δh of 100 miles and Δv of 6000 fps.[12] Additional 1% error in burnout time and a 30° angular altitude error[13] for a 1° thrust misalignment in a Δh of 100 miles would be imposed if these errors were present during the descent.

5.72 A study of requirements and expected errors was undertaken for a fully automatic system intended for unmanned operations. The problem undertaken was for the Surveyor Program, but the conclusions are pertinent to the Apollo task. Some design requirements for a variable thrust system were generated; it called for a thrust variation of 50 or 100 to 1. Thrust variation state-of-the-art is indicated to be 25 to 1 with moderate development; possibility of 50 to 1 ratios with considerable development in both injector controlled and variable throat area combination models. The X-2 aircraft varied thrust from 5 to 1 by means of two chambers in 2500 lbs steps from 2500 lbs to 12,000 lbs thrust. The X-15 aircraft also possesses variable thrust but the specifications are not known.

5.73 A two stage deceleration scheme was devised [14] and considered feasible. It consisted of a fixed thrust first stage and a variable thrust second stage. This arrangement reduces the need for the maximum thrust variation to about 20 to 1.

10/ Project Apollo Statement of Work - Phase A, NASA, 28 July 1961, CONFIDENTIAL.

11/ Astrionics Lab Project 9(632-5215) for Wright Air Development Center, April 1959, CONFIDENTIAL.

12/ Ibid

13/ Neglecting additional altitude control such as reaction jets.

14/ Astrionics Lab. Project, op. cit.

5.74 System requirements[15] were generated for a small two-stage landing system of acceptable performance in terms of available or short range capabilities.

5.75 The hardware for lunar landing is probably capable of being built to the performance specifications required within the allowable time frame of 5-7 years. But, it is perceived that major problems are inherent in the electronic control sensors in the closed loop system. Of critical effect to the control mechanisms is the possibility of positive or negative bias in the altitude, velocity or vector sensors. For instance, positive or additive bias of the altitude input would cause impact before velocity approaches zero and subtractive bias would cause premature cutoff resulting in free fall to the surface. In the previously cited study, the maximum system bias error for satisfactory landing of a medium sized unmanned vehicle is calculated for the second (variable thrust) stage. The landing trajectories can differ in terms of the maximum retro g-force as the vehicle slows above the lunar surface. Maximum bias error in landing system for 1.5 and 3 g are shown in Table 23.

TABLE 23

MAXIMUM BIAS ERROR IN LANDING SYSTEMS

Parameter	Bias 1.5 g	Bias 3 g
Sensor Requirements		
1. Thrust acceleration	-1.5 to + .05 fps^2	-6 to + 2 fps^2
2. Thrust direction	$\pm 1^{\circ}$	$\pm 1^{\circ}$
3. Altitude determination	-2 to + 4 ft	-2 to + 1.5 ft
4. Velocity determination	-5 to + 3 fps	-5 to + 6 fps
5. Velocity direction	$\pm 1^{\circ}$	$\pm 1^{\circ}$
Computer Requirements		
1. Δ altitude req'd	.01 fps^2	.01 fps^2
2. Thrust acceleration req'd	.1 fps^2	.1 fps^2

5.76 Work is continuing on the feasibility of providing a closed loop with man included. In view of recent orbital performance and a lot of simulated mission results, manned control could be of major value and considered thoroughly in terms of the lunar landing sequence. Success in this concept is

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15/ Velocity Control System Requirements, ITT Report 2057, WADC Weapons Guidance Laboratory, April 1959.

indicated[16/] until about 400 ft altitude was reached, then the early simulations were suboptimal and erratic. A switch to two-pilot technique splitting the altitude control and closing velocity tasks resulted in very successful results; generally less than 4 fps residual velocity and within 2000 ft of a prescribed point of contact. Continued intensive practice in the one-man mode after a time showed no appreciable difference in results from the two-man mode including total fuel consumption.

5.77 In summary, it is felt that the control hardware for throttling and gimbaling for lunar landing either direct flight or from orbit will be able to be developed to the degree required. But, most problems will be associated with the environmental effects on the reliability of this hardware and the accuracies or bias of the electronic techniques generating the guidance inputs for the control systems.

Lunar Surface Launch

5.78 The launch from the lunar surface will be an example of one of the most sophisticated machines ever designed; launched from the most primitive and adverse surface environmental conditions.

5.79 As a control problem, the task is not particularly prohibitive. The launching ascent, once the surface is cleared, is not subject to atmosphere disturbance and the control problems akin to it. The space vehicle launched from the lunar surface will be cylindrical, about 30 or 40 ft long, 14 feet in diameter, and weigh around 35-45,000 lbs. This is generally a bit larger, particularly in diameter, than the present Polaris IRBM.

5.80 Bending and stability problems at launch for such a configuration would be almost negligible. The launch ignition, early recovery and ascent would be reasonalbly identical to the Polaris launching; being initiated under alignment of something less than vertical and without the benefit of sophisticated radio communication from ground support complexes. The Polaris control system is capable of re-orienting its flight from launch angles up to 30° from the vertical. Since the Polaris gimbal angles are no greater than those designed for the lunar launch module, this unimproved launch site problem seems to be within the predicted capability. Of course, there are some inconsistencies in the comparison such as the fact that the Polaris launch is characterized by a small initial upward velocity at motor ignition and the solid engines have high thrust rise tendencies producing high control torques very soon after ignition. The lunar takeoff module should be capable of alignment to the vertical within ± 10° and the large angle correction capability would

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16/ Aviation Daily, Volume 183, No. 19, 16 January 1962, Laboratory source unknown.

be required only as emergency mode control with the required levels of correction arrived at through logical probability techniques.

5.81 The same early ascent characteristics will be present in either a lunar orbital injection from the surface or a direct trajectory return to earth.

5.82 With all atmospheric and staging type disturbances removed from the flight, the control systems are relegated to steering and attitude drift functions which are not construed as major problems in any way. Development of systems should be of engineering nature adopting the present techniques and feedback circuits to the vehicle configuration used in the lunar launch sequence.

5.83 The same comments are applicable to this mission sequence as the landing sequence. Namely, that the required hardware and its desired performance are attainable in terms of present capabilities, but that a critical area of interaction is the time-environment reliability effects upon the components of the control system, particularly the gimbal hinges and hydraulic components.

5.84 The accuracy characteristics of the mission sequences are covered thoroughly in the guidance section and could be considered as requirements for the guidance and control systems.

5.85 The true accuracy requirements of a gimbal actuator system are hard to isolate because of the presence of the closed loop feedback techniques. Hardware induced error is damped out by its own manifestation through the closed loop system.

5.86 This is a favorable situation in the cases where hardware error can arise through the environment time effect. But, due to the rigid fuel and weight constraints on the later acting stages of the space craft[17] where the error is more prone to arise, there is not much room to include extra capacity for negating or damping possible errors. For this reason, the time-environmental-reliability effects on control subcomponents remain a problem and their attenuation should remain a high priority goal.

5.87 The surface launch trajectory will be followed by orbital launch and/or midcourse correction sequences as the vehicle returns to the earth. These sequences are relatively the same as the earth outbound trajectory and those discussions will suffice for the inbound trajectory until the re-entry sequence is reached.

17/ 13.3 lbs added launch weight from earth for every 1 fps midcourse correction of a 15,000 lb capsule.

Reentry

5.88 Capsule reentry will terminate the lunar flight whether aborted or successful. Except in the case of very early booster ascent abort, the abort and recovery systems will be the major operating system and the reentry protection components will play almost a negligible role.

5.89 The basic control problem in reentry is in two parts 1) to control the entering speed of the capsule by means of some energy dissipation scheme whether it be retro techniques, atmospheric "skipping," drag, or some combination of these and 2) to provide some terminal control when possible to enhance the chances of landing safely in a preselected recovery area.

5.90 At the present time, the Mercury and Apollo capsule reentry schemes are primarily concerned with the solution to the first problem and much work is being accomplished toward this end. It has been generally computed considering a fixed volume, maximum entry corridor, minimum weight capsule, that a maximum L/D ratio of about .5 is desirable.[18/]

5.91 These low L/D configurations are generally dependent on the angle of attack during reentry to give the desired drag-ablation combinations for safe reentry.

5.92 Control for these maneuvers consists of an impulse to change vehicle angle of attack quickly and accurately. These maneuvers will be initiated at around 75-100 miles. It is desirable to hold peak g loading below 14 g in the eyeballs-in direction if the pilot is required to monitor and backup the entry configuration control systems.[19/]

5.93 The method of providing the control impulse is generally considered to be the use of aerodynamic surfaces after the capsule enters atmospheric levels which are dense enough to provide adequate dynamic pressure. Early reentry attitude can be brought about by the reaction jet attitude control system while the capsule is on the outer fringes of the detectable atmosphere.

5.94 Particular control problems which are presently being evaluated are roll-yaw cross coupling, lateral and directional stability, general aerodynamic stability through a range of angles of attack and Mach values.

18/ R. W. Rainy, Summary of Aerodynamic Characteristics of Low Lift-Drag Ratio Reentry Vehicles from Subsonic to Hypersonic Speeds, NASA TM X-588, September 1961, [redacted]

19/ Cree, Brent and Douvillier, Influence of Sustained Accelerations on Certain Pilot Performance Capabilities, Ames Research Center, July 1961, [redacted]

5.95 The stability problems seem to be resolving into minor problems with the exception of the roll control-yaw cross coupling effects, and the oscillatory modes that can occur if damper fins fail during critical stages of the reentry maneuvers. These particular problems occur in some of the shapes tested in hypersonic wind tunnels. Results [20] with these and other shapes indicate that with proper modifications; blunt, low L/D vehicles can be made aerodynamically stable and controllable at angles of attack which encompass zero lift, maximum lift, and maximum L/D ratio. Three of the configurations studied (cant nosed, flat bottomed, half cone; connex-faced body of revolution; and blunted cone) required only minor modifications to obtain satisfactory aerodynamic characteristics.

5.96 As in the case with fluid dynamics, it is possible to compute with reasonable confidence some of the basic body characteristics; however, such is not possible with body shapes possessing rapidly changing surfaces and various results from edge relief, or carryover effects.

5.97 In placing controls to provide satisfactory aerodynamic characteristics, the type and location of the controls must be selected with care to avoid problems such as cross coupling. At present, in the computation of control effectiveness, it is seldom possible to theoretically account for local conditions and flow phenomena in the vicinity of deflected controls, [21] and the reliance upon experiment is mandatory in the case of control configurations for low L/D reentry capsules.

5.98 If advanced work reverses the current conclusions concerning aerodynamically stable shapes and successful attenuation of cross control effects, some indications are available [22] that required piloted reentry maneuvers can be performed without any aerodynamic controls by using vertical center of gravity offset to trim at the required L/D ratio and use reaction jet controls to make rolling maneuvers. In regards to this scheme, it does not seem a good solution to attempt cg adjustment in the capsule while under high g loadings. There would seem to be inherent structural and response problems in moving masses of sufficient weight to shift and stabilize the cg appreciably, within relatively short time frames, to result in changing angle of attack up to 75°.

20/ NASA TM X-588, op. cit., [redacted].

21/ Boxer, Rainey, & Fetterman, Aero-Dynamic Characteristics of a Variety of Low-Lift Drag Ratio Reentry Vehicles, Langley Research Center, July 1961, [redacted].

22/ Moul, Schy, and Williams, Dynamic Stàbility and Control Problems of Piloted Reentry from Lunar Mission, Langley Research Center, July 1961, [redacted].

5.99 There is not much opinion available as to the merit of a 100% attitude jet system under the conditions associated with direct reentry from the lunar trajectory. It would seem that reaction jet control would be a feasible alternate or redundant control system. Certainly it was proven as a workable system during piloted orbital reentry in the recent manned flight. However, there are admittedly more adverse conditions in escape velocity reentry than are present in orbital reentry.

5.100 The presence of higher heat exchange rates in the lunar return could pose control actuator problems. The control surfaces will be prone to high levels of localized heating. As could be expected, the extent of localized heat transfer is large for large flap deflection angles.[23]

5.101 There seems to be no solution offered to the problem of control surface heating except statements that the extent to which flap heating can be minimized is dependent on trade-off studies concerning aerodynamic characteristics of the particular system approach employed.[24] It may be assumed that if the flap heating problems are insurmountable, that attitude jet controls may be adjusted for angle of attack control during the reentry sequences.

5.102 The discussion of the winged type capsule is not undertaken in detail at this time because the higher weight penalties[25] involved in the higher L/D ratio vehicles are considered prohibitive in terms of the lunar landing-return mission booster capabilities at this time. The other control problems of winged capsules would be generally parallel to the previous discussion on low L/D capsules.

5.103 The problem of controlling the capsule to a chosen landing point is not really considered in most schemes. The recovery is usually effected by attempting to predict the approximate landing area and cover it with mobile recovery forces located at strategic positions. As long as low L/D capsules are considered and terminal landing is by means of parachute devices, the problem will remain one of controlling the recovery forces toward the landing point and not attempting landing point control in terms of location of recovery forces.

5.104 In summary, the basic stability of the capsule does not seem alarmingly difficult to attain according to recent works. However, the problem of adjusting the angle of attack (i.e., the L/D ratio) of the capsule could be difficult by

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23/ Stainback, Jones, and Coe, Convective Heating of Basic Shapes for Lunar Mission Vehicles, Langley Research Center, July 1961, CONFIDENTIAL.

24/ Ibid.

25/ For equal volume and payload capacity capsules.

reason of high localized heat rates in areas of control flap deflection. These heat levels could not only affect the surface structural material but also can affect the reliability of the actuator systems if thermal conductive rates allow high BTU input to fluids, seals, O rings, or lubricated bearing surfaces. Attitude jet systems may be employed if established to be reliable at the higher dynamic pressures and heat rates associated with direct lunar trajectory reentry.

5.105 The winged capsule configurations capable of landing spot choice are too heavy for the lunar landing mission. Until the capsule weight requirements allow such configurations, the earth landing recovery problem will still be the strategic location of highly mobile recovery forces in patterns consistent with predicted impact areas. There are possible minimal steering techniques available for parawing or parachute components which appear to be feasible if desired.

PART II. FLIGHT DISTURBANCE CONTROL

Engine Out

5.106 The capability to control the disturbances of engine malfunction currently depends on cutting the bad engine out of action in the cluster before the malfunction brings catastrophic explosion. Therefore, the major flight path disturbance is not from the cause but the cure.

5.107 The engine-out control technique has been devised to monitor critical engine values (speed, chamber pressure) and initiate a cutoff signal if the values rise or fall more than the expected amount. This effectively raises the cluster reliability in terms of mission success because it removes the failure threat associated with uncontrolled engine malfunction resulting in catastrophic occurrence. Reliability results with and without the engine-out control are shown in Table 24.[26/] The chances of initiating a false cutoff signal are included in this reliability table.

TABLE 24

NOVA CLUSTER RELIABILITY FOR SUCCESSFUL 1st STAGE BOOST

	1 F-1 Engine	8 F-1 Cluster	8 F-1 Cluster With Engine-Out Systems
Predicted	.995	.961	.988

5.108 The loss of an engine during first stage boost introduces two problems: (1) the mission performance capability is reduced by loss of thrust, (2) the controllability is affected by the thrust imbalance of the remaining engines, necessitating gimbaling, and if the failed engine is a control engine, the control torque available to compensate for later perturbations is reduced.

5.109 The second problem can be limited in seriousness by proper design of sensing components, feedback loops, and available gimbal control capacity. But there is currently no method to replace the lost thrust of an engine-out disturbance.

5.110 There are two alternative solutions as long as there is no possibility of replacing lost thrust: (1) demand high cluster reliability and

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26/ North American Aviation, Report SID 61-327, October 1961, [redacted] lift capacity payloads; or (2) accept lower cluster reliability and lift a lighter payload whose weight will still allow mission success with N-1 engines should malfunction occur. Table 25[27,28] relates the payload capacities in lbs for the 4 engine and 8 engine vehicles for orbit or escape missions with or without one engine gone.

TABLE 25

PREDICTED PAYLOAD LIMITS

VEHICLE	TO ORBIT		TO ESCAPE	
	N Engines	N-1 Engines	N Engines	N-1 Engines
C-4 (N = 4)	220,000 lbs	Successful mission possible. Unknown payload	96,000 lbs	Marginal for mission failure
NOVA (N = 8)	370,000 lbs	315,000 lbs	183,000 lbs	150,000 lbs

Certainly, an engine-out system is of dubious need if an engine failure, catastrophic or not, results in mission failure. One method of engine-out control would be effective payload weight scheduling to allow automatic choices of alternate missions should an engine failure early in the program nullify the chances for success of the intended mission.

5.111 Alternate missions should be capable of completion with 1 engine out. This problem is important and should be fully understood before Apollo mission scheduling is frozen. Of prime importance is a method of uprating remaining engine thrusts upon single engine malfunction in order to complete the intended mission with as large a payload as possible.

5.112 Analysis has shown that the second disturbance (thrust unbalance, wind controllability) is not evidently serious in terms of design specifications for the F-1 engine, the C-4, and Nova vehicles. Table 26 shows the results of calculations to determine the disturbance from a malfunctioning engine operating at 50% of capacity thrust for a two second transient period. The two second period is picked arbitrarily but it is highly possible that the engine could malfunction to 50% thrust in 1 second until a

27/ Lockheed Georgia Co., Report ER 5388, October 1961, [redacted]

28/ General Dynamics/Astronautics, Report No. AE 61-0967, October 1961, [redacted] a cutoff decision, then decay to negligible thrust in .7 to .9 seconds,[29] resulting in about 50% average thrust for the 2 second time frame. The gimbal requirements are given for zero wind and 2σ wing (74 meter/sec)[30] at Cape Canaveral, Florida. Wind effects are for a non-winged payload at peak dynamic pressure at first stage boost. Similarities in the C-4 control and non-control engine-out columns are because the four C-4 engines are all gimbaled and any failure is a control engine failure.

TABLE 26

PREDICTED ENGINE-OUT GIMBAL REQUIREMENTS

Non-winged payload
peak dynamic pressure
2 second, 50% thrust unbalance

N engines
N' control engines

VEHICLE		CONTROL ENGINE OUT	NON CONTROL ENGINE OUT	ALL ENGINES IN
C = 4 N = 4 N' = 4	2σ wind	3.97°	3.97°	2.15°
	No wind	1.17°	1.17°	0°
NOVA N = 8 N' = 4	2σ wind	3.81°	2.86°	2.36°
	No wind	$.65^{\circ}$	$.495^{\circ}$	0°

It will be noted that the no-wind gimbal requirements would almost be considered negligible since the RMS uncertainty of the F-1 gimbal system is reported to be $.59^{\circ}$ plus $.4^{\circ}$ for snubbing action.[31] However, this amount of uncertainty does demand a reliable closed loop system.

5.113 Since the thrust unbalance portion of the gimbal requirements in Table 26 are negligible, Figure 20 has been included to show the time growth of a 50% control engine-out situation uncorrected. Figure 20 is a simple rigid body rotation curve with no aerodynamic effects considered. It serves to show that reaction time is not particularly sensitive for the first

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29/ Ibid.

30/ Ibid.

31/ Ibid.

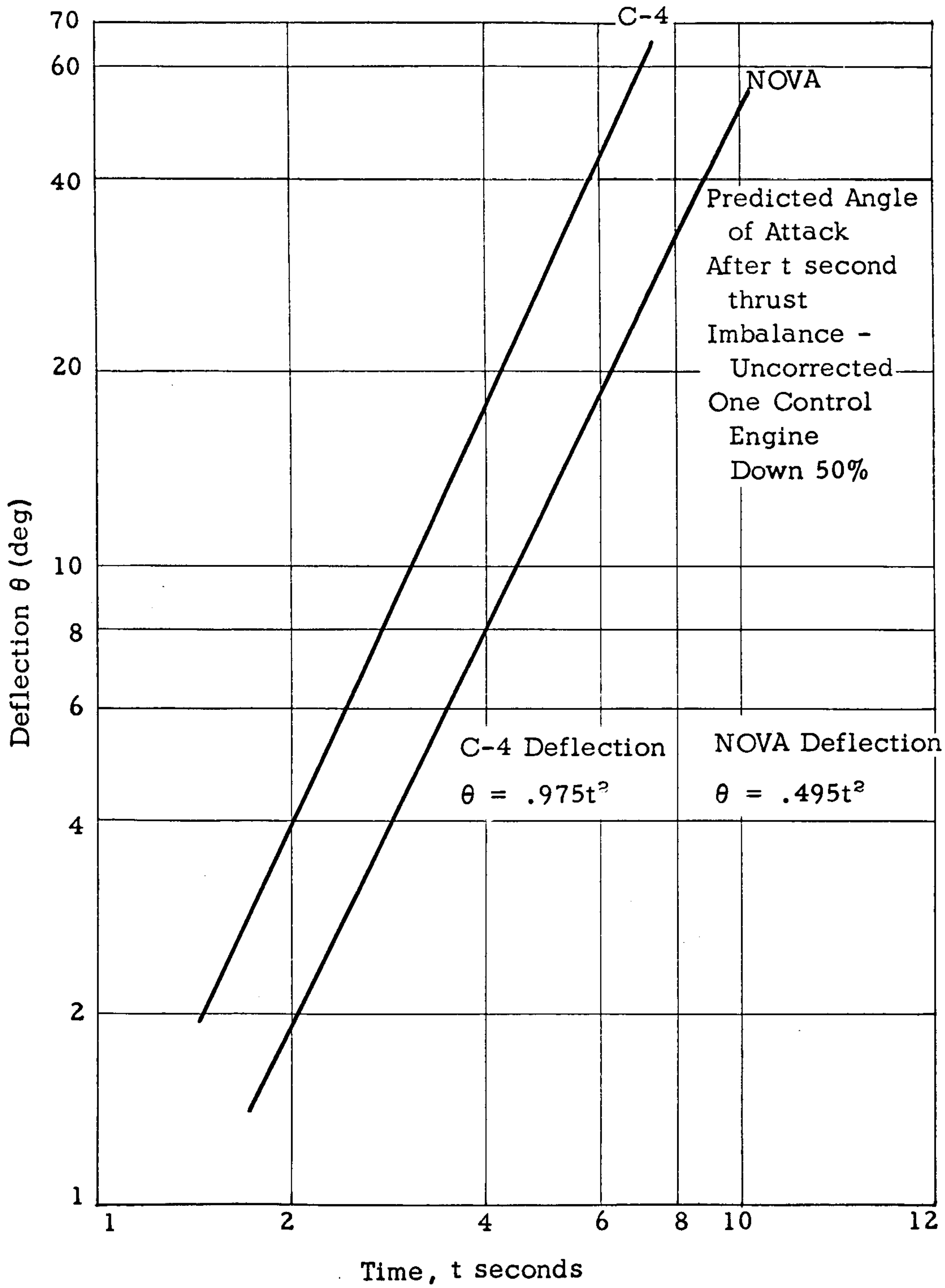

FIGURE 20. TIME GROWTH OF A 50% THRUST CONTROL ENGINE-OUT SITUATION

2 or 3 seconds after an engine-out shutdown. The 6° gimbal (8.4° on diagonals) capacity of the F-1 engine seems to hold adequate for these disturbances if they are corrected within 2 or 3 seconds.

5.114 Second stage engine-out capability is subject to the same disturbances as the preceding first stage discussion except that as the stage rises above the atmosphere, windshear and dynamic pressures drop to negligible values. The J-2 cluster in the S II stages have similar gimbal capacities (6° which are considered adequate since second stage disturbances will probably be less severe than initial boost. Table 27 documents the value of the aerodynamic forces about the center of pressure by available disturbance control moments about the center of gravity. U_{α}/U_{δ} serves as a predicted proportionality between gimbal angle and controllable steady state angle of attack

$$\left[\begin{array}{l}\text{required}\\\text{gimbal angle}\end{array} = \frac{U_{\alpha}}{U_{\delta}} \ \text{angle of attack}\right] .$$ [32,33]

It is possible to see from Table 27 that the one engine out does not seriously hamper 2nd stage controllability. Current information has not been found concerning mission completion capabilities with one S II engine lost.

TABLE 27

S II STAGE CONTROLLABILITY
ONE CONTROL ENGINE-OUT 100%

	S I	S II
C-4	.30	.041
NOVA	.33	.0063

Control Malfunction

5.115 Control component malfunction would creat disturbances stemming from causes such as an engine stuck at maximum gimbal, an engine swinging about in an unconstrained manner, a failure to respond to computed

[32] Ibid.

[33] Part I, NASA Industry Apollo Technical Conference, July 1961, [redacted]

thrust vector requirements, or some electrical mismatch of guidance signals to servomechanisms. Malfunction in any control system is bound to result in some type and magnitude of disturbance.

5.116 It seems feasible to consider that only one hydraulic gimbal system would fail at any time since computed reliability of the F-1 gimbal unit is reportedly .999.[34]

5.117 If one engine erroneously is at maximum gimbal angle θ; then the N useful remaining engines swivel

$$\frac{\theta}{N-1} \text{ degrees}$$

to counteract the disturbing torque. The longitudinal thrust component remaining is then

$$T \cos \theta + (N - 1)\left(T \cos \frac{\theta}{N - 1}\right)$$

where

T = single engine thrust

N = number of engines

θ = single erroneous gimbal angle

for

$\theta = 6^\circ$ (maximum).

Results of the calculation are shown for the 4, 5, and 8 engine vehicle in Table 28.

TABLE 28

PERCENT LONGITUDINAL THRUST REMAINING AFTER 6° CONTROL MALFUNCTION IS CORRECTED BY GIMBALLING REMAINING ENGINES. S I STAGE

NO. ENGINES	% REMAINING
N = 4 (C-4)	99.81%
N = 5 (C-5)	99.86%
N = 8 (NOVA)	99.92%

[34] North American Aviation, Report SID C1-327, October 1961.

5.118 The problem of any malfunctioning vector angle control apparatus is not critical insofar as one-engine events are concerned. The control may take the form of locking the engine in any vector (even hard steer). The disturbance will reach a steady state and be counteracted through the closed loop system aligning the remaining engines while still using the longitudinal vector of the stray engine for useful work in completing the mission.

5.119 The small percentages of diverted thrust are negligible for primary boosting in first or second stages. But, for the long trajectory injection toward the lunar target it could prove more serious.

5.120 In order just to hit the moon at any point with an uncorrected trajectory, the velocity error must be held within 75 fps in 36,000. This accuracy may vary some amount relative to the accuracy of other launch variables, but may be well represented by this figure.[35]

5.121 If, at orbital launch, checkout showed an inactive or stuck hydraulic gimbal system, the abort decision would have to be based on the capability to achieve the desired velocity accuracy with the degree of possible thrust diversion.

5.122 Since the trajectory and midcourse corrections do not require much gimbal capacity, minimum angular requirements for this sequence would negate the seriousness of larger thrust diversion from actuator malfunction and help enhance the capability to reach the desired accuracies of velocity.

5.123 Upper stages of vehicles that are required to remain in orbit before ignition will be particularly subject to hydraulic actuator sluggishness or failure because of fluid viscosity increase or even solidification. Several heating methods are possible ranging from orbital orientation for solar heating to on-board electrical heating jackets.

5.124 The reliability of an unprotected hydraulic system in a cryogenic environment will fall rapidly. Table 29[36] shows some expected orbital stay times for the lunar mission for the various vehicles. The smaller vehicles stay longer because there must be more rendezvous flights to assemble and outfit the orbital launch vehicle and space craft prior to launch.

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[35] cf. Section II, Astronautics.

[36] Lockheed Georgia, Report ER 5788, October 1961, [redacted]

TABLE 29

EXPECTED ORBITAL STAY TIMES

VEHICLE	POSSIBLE ORBIT TIME BEFORE LAUNCH
NOVA	12 hrs
C-4	68 hrs
C-3	212 hrs

5.125 The possibility of mechanical gimbal systems to overcome the environment problem seems feasible since large angular capacity is not required and the attitude jet systems are capable of good attitude orientations prior to thrust periods.

5.126 The most critical component failure would be an error in the servo circuits that translate the guidance signal to engine motion. This type of malfunction is uncontrollable and shows the need for control computer and servo reliability to insure missions success.

5.127 All phases of the Manned Lunar Flight are subject to engine and control system disturbance. To control the disturbances seems to be within our present capability. However, the major project is still to prove component reliability as a first defense against the disturbances and secondly, to involve a reserve thrust capacity to replace lost thrust due to engine or control malfunction.

Wind

5.128 Wind shear has been introduced in the discussion on engine-out capability. Extremely bad wind conditions are sufficient reasons to halt vehicle launches. The problem of determining actual wind environment serves two important purposes: (1) to provide data for design criteria, and (2) to determine the relative probability of success just prior to a planned launching.

5.129 Some work has been done[37] on smoke trial analysis and is showing improvement for these purposes over the current balloon sounding method. The preceding discussions have shown that design requirements for C-4 and NOVA will exceed the needed capabilities for 2σ winds (75 m/sec) at Cape Canaveral even with one control engine not functioning. However, superior as this might be, added controllability

[37] H.L. Runyan and A.G. Rainey, Launch Vehicle Dynamics, NASA Industry Apollio Conference, Part I, July 1961, [redacted]

reserve will be insured by placing the gimbal diagonals in the vehicle pitch plane thereby taking advantage of the increased gimbal angle ($\pm 6^{\circ}\sqrt{2}$) on the diagonal.

5.130 The available wind moments were calculated for the symmetrical payload of the launch vehicle and Apollo space craft on top of the booster. The launch of any winged payload will considerably change the wind moment on the space craft. A study should be undertaken to determine whether the wind force vector from the winged payload should be in the pitch plane to take advantage of more gimbal angle; or whether the vector should lie in the yaw plane to take advantage of less wind velocity in the non-prevailing wind directions. (Table 30).[38]

TABLE 30

CAPE CANAVERAL WIND

Prevailing Wind	Non-Prevailing Wind
W → E	N → S
	S → N
	E → W

5.131 In terms of transverse winds, the winged payload, being on the very largest lever arm possible will raise considerable disturbance on initial boost. The gimbal angle capacity required to counteract such disturbing moments is variable proportional to the disturbing moment

θ = gimbal angle required

α = angle of attack at time t after onset of disturbing moment

$F_{\alpha}d_{\alpha}$ = aerodynamic disturbance moment

$F_{c}d_{c}$ = available control moment

$$\theta = f(\alpha)$$

$$f(\alpha) = \frac{F_{\alpha}d_{\alpha}}{F_{c}d_{c}}(\alpha)$$

but $F_{c}d_{c}$ is severely limited and can be considered relatively constant.

$\therefore$ θ directly proportional to $F_{\alpha}d_{\alpha}$.

38/ General Dynamics, op. cit.

5.132 It can be seen that this problem area could severely tax the gimbal capacities for low wind values directly against the broad surface of a winged payload on the tip of the earth launch vehicle. Not only first stage control is involved in the winged payload problems because staging occurs while still in the atmosphere and the wind-caused angle of attack value is critical for successful staging. Wind values predict a conservative angle of attack at staging of 3°[39] for the Apollo capsule. For simulated runs, C-3 vehicle, Apollo payload, and attack at staging equal to 4° from expected winds; the resultant angle of attack was 10° at S II ignition after short coasting. To correct this, the S II engines were at full gimbal (6°) for 2.5 seconds.[40]

5.133 There are basic aerodynamic problems associated with winged payloads that should be resolved before attempting to adopt Apollo boosters to winged vehicles.

<u>Staging</u>

5.134 A staging of the launch booster occurs in the atmosphere for all possible lunar flight tests, probes, or missions and is one of the most critical periods in determining a successful flight. The C-4 and NOVA vehicles both stage at about 2×10^6 ft. At this altitude all the preceding disturbances discussed have had a chance to contribute summarily to the flight path at staging.

5.135 A second set of influences on the relative success of staging are the parameters of the vehicle itself and certain constants present in any separation-ignition technique. The controllability of these disturbances is manifested in three areas: (1) ability of S I to deliver S II to staging altitude and cease thrust with a minimum dispersion (angle of attack) about the flight axis, (2) the ability of S II to coast for the pre-ignition period with little or no increase in the angle of attack, (3) the ability of S II to correct what dispersions are introduced during the staging process and S II ignition.

5.136 The controllability is subjected to various constraints such as thrust decay and build rates, rocket flame impingements at separation, ullage requirements and engine-out capabilities. Successful simulations

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[39] General Dynamics, <u>op.cit</u>.

[40] The C-3 vehicle (2-F-1 engine) possesses the worst staging characteristics of the Saturn series.

were run on the 2-F-1, C-3 vehicle which has much worse staging characteristics than either the 4-F-1, C-4 or 8-F-1, NOVA.[41]

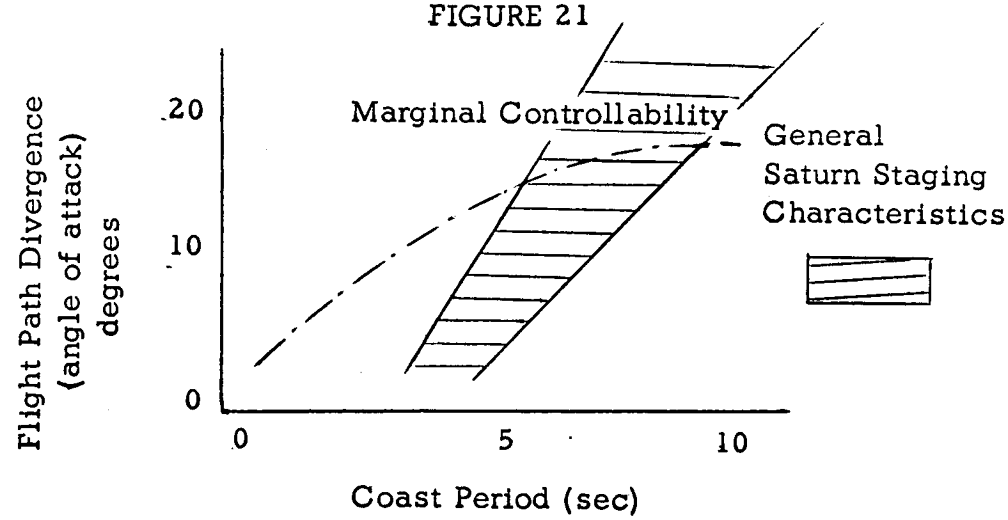
5.137 The results will not be discussed in this section as the C-4 vehicle is considered as submarginal for the current approaches to the lunar mission. Instead, the several variables that can increase probability for successful staging will be discussed and their requirements on the control systems will be shown.

5.138 Rocket flame and blast impingement at separation dictate a period of uncontrolled coasting separation for the two normally unstable vehicles. During this period of short coasting, the vehicle unstability gives rise to tumbling impulses that effectively change (increasingly) the angle of attack in relation to the direction of flight. Figure 21 shows the approximate limits for the coasting-flight angle.

5.139 As the angle of attack at ignition grows, the required gimbal angle on S II for return to flight path grows, and as the possible gimbal angle requirement at ignition grows, the chance of high normal control forces stressing the structure and bending moments are involved.

5.140 There are some feasible techniques[42] which are applicable for the attenuation of the coasting-separation in the Saturn-type vehicles.

FIGURE 21



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[41] General Dynamics, op.cit.

[42] Ibid.

They are:

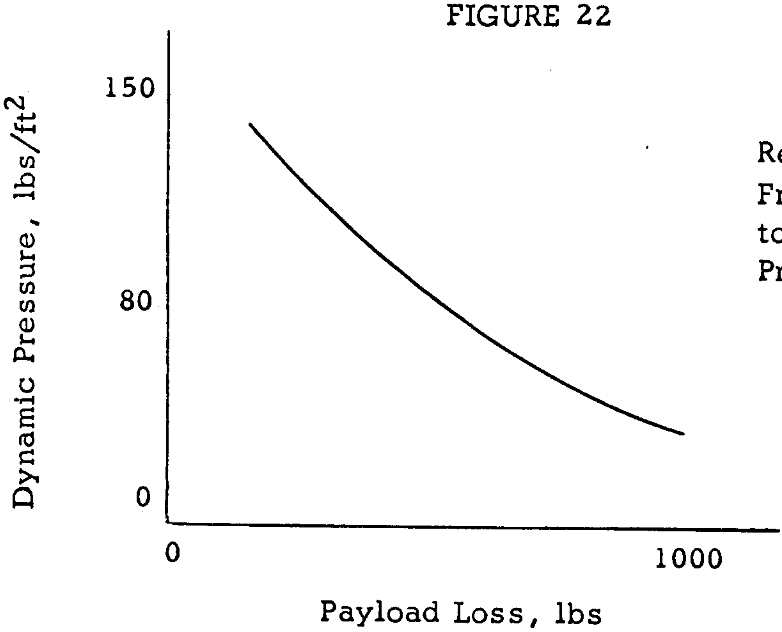
a. Loft the trajectory higher for staging.

b. Change the stability characteristics.

c. Addition of control during coasting.

The technique to allow less dispersion during coast by lofting the initial trajectory to allow smaller dynamic pressures at the staging velocity is feasible because it is one solution that does not tend to reduce reliability. The most significant tradeoff is loss of payload capacity for lofted trajectory as seen in Figure 22.

FIGURE 22



Resultant Payload Loss From Lofting Trajectory to Escape High Dynamic Pressures at S-I Staging

5.141 The inertial properties of the vehicles (particularly S II) can be changed to provide a more stable aerodynamic moment for uncontrolled coasting. This action results in minor structural and fuel feed complications which reduce payload through added structural weight by about the same predicted magnitude as the lofted trajectory, but with a drop in some system reliabilities—mainly propellant feed.

5.142 For addition of attitude control during coast, heavier S I retro-rocket systems or canard surfaces all result in better control of angle of attack during coast but penalize payload and introduce lowered mission reliabilities by virtue of added components.

5.143 Some study has shown that S II liquid engine ignition is not impaired by fuel floating in tanks during very short periods of zero gravity coast.[43] This particular constraint should be thoroughly understood before such design considerations are actually planned for.

5.144 All the control information presented in this discussion is considered as magnitude prediction only and indicative of information and analysis that should be firmed in order to make efficient decisions regarding final configurations and design parameters for the manned lunar shots.

5.145 In retrospect, staging is a critical area, the Saturn flight plans show 2 atmospheric stagings required for each orbital lunar shot. The sequence is a concentration of many adverse effects and there exists fine lines between mission success and failure. Three more separation phases occur per lunar mission but these are basically unpowered jettisons that do not contribute such disturbances as they occur above the atmosphere and in the vicinity of the moon. The unpowered stage jettisons will change the dynamic moments and characteristics of the space vehicle. These changes will affect the vehicle reaction to attitude pulses and increase the sensitivity to any thrust misalignment at later sequences.

Structural Vibration (Bending)

5.145a With the advent of large high performance missile and space boosters, the light weight flexible airframe, and its related control system—structural instabilities have become important design problems in new booster development.

5.145b Present airframe structures are subject to dynamic phenomena which are usually undesireable. Missile flexibility is clearly recognized as a property which can adversely effect the control system sensors; the major flexure disturbances which affect control systems are the first three modes of long body bending and the first torsional mode in twisting.

5.145c Within the vehicle, there is a guidance and control system that controls the orientation of the engine thrust vector. Angular rate, including flexible body bending, are sensed by gyros which transmit corrective signals through the flight computer to the servo-hydraulic mechanisms. The vehicle respons to these signals by vectoring the engines, which initiate further elastic deformations and these in turn produce additional gyro signals. This interaction or coupling is the basic feature of structural feedback. This feedback interaction will, if uncontrolled, provide such flight path disturbance that the vehicle can oscillate destructively.

43/ General Dynamics, op. cit.

and the flight have to be aborted. With the flexible airframe of the present boosters, the bending must be tolerated and the only real method of disturbance control is to prevent the initiation and propagation of harmonics leading to destructive oscillation.

5.145d The current method is to pick a control frequency as far as possible from the natural vehicle bending modal frequencies and at the same time provide effective electronic filtering of the structural feedback signal from the rate gyros to the guidance system. The basic success of the current method rests not primarily on the control systems, but upon the analysis of the vehicle dynamics which serves as the primary input to the control system stability parameters.

5.145e The analysis of dynamic flight response of a missile involves consideration of factors which are often perculiar to the specific booster configuration. This consideration alone limits the usefulness and/or accuracy of any single method of analysis. However, in spite of the presence of enumerable variable in bending and torsional analysis; the results[43a, 43b] of many flight tests of all types of flexible boosters show that the present analytical methods are sufficient to predict and isolate successfully the dangerous instabilities prior to flight testing. The majority of in-flight oscillations that are present in early flight testing are not destructive and are relatively easy to eliminate or attenuate as control inputs.

5.145f Bending mode stability may be achieved throught the control system in two ways: attenuation stabilization and phase stabilization. Attenuation stabilization holds fewer in uncertainties and is the most favorable method when the magnitude of the bending mode frequency is well above the control frequency. Most large boosters have the first bending mode close to the control frequency and therefore, must resort to phase stabilization for that mode because straight attenuation of the filter interferes with the normal gain values. Phase stabilization consists of procedures whereby the filter is designed to include lead and lag units to produce maximum stability for a prescribed gain level. The design and development of these electronic filter networks is well known and should pose no major problem in Saturn series or Nova design and development.[43c]

43a/ Waymeyer and Sporing, An Industry Survey on Aeroelastic Control System Instabilities in Aerospace Vehicles, IAS paper 62-47, Jan. 1962.

43b/ Miller & McLauphlin, Summary of Flight Data of Loads Significance for Five Types of Large Missiles, NASA TMX-510, July 1961, [redacted]

43c/ The first Saturn flight, SA-1, (Oct. 27, 1961) was successful and showed good compliance between predicted dynamic response and actual control stability (NASA TRX-500 (?)).

5.145g An adjacent problem to filter analysis is the choice of location for the rate gyros in the airframe. This decision also is based upon the bending analysis performed before flight testing. Positions along the longitudinal axis are usually chosen and the characteristic of the bending at that point is taken into consideration in deciding the techniques to provide the required dynamic stability during the flight. The positions of the bending nodes and antinodes and their relative movement up or down the longitudinal axis during flight progress is a factor of consideration in the location problem. These analyses also should cause no major problems in large booster design and development.

Sloshing

5.145h Sloshing and other oscillatory disturbances due to the unrestrained motion of liquid propellants in a large space booster are controlled in much the same analytical manner as the structural bending problems of the previous section.

5.145i The severe sloshing-interaction frequencies are isolated by initial calculations, then the control rates are adjusted to avoid severe excitation of the liquid-tank systems. The feedback circuits are sometimes filtered electronically to avoid the generation of erroneous flight commands due to sloshing-structural-control instability.

5.145j In the case of the first flight test of a Saturn type booster, the criteria for designing the filters neglected the effect of the sloshing propellant.[43d/] This is permissable when correlated with the assumption that forces resulting from propellant sloshing will not have appreciable effect upon vehicle bending modes and control stability. The assumption is valid as long as the frequencies are well defined and no dependency is made upon the control system to provide damping of the sloshing propellant. When the control systems are divorced from slosh damping, the control of the liquids must be achieved by the incorporation of anti-slosh devices into the propellant tanks. Weight is a critical constraint in vehicle design, so damping devices such as baffling must be designed to provide acceptable stability with minimum weight.

5.145k Acceptable slosh suppression was gained in the SA-1 flight by the addition of Z-rings in the upper parts of the clustered circumferential 70" tanks and no dampers at all in the central 105" tank.

43d/ Robert S. Ryan, Control Flutter Stability Analysis of Saturn SA-1, NASA TM X-400, January 1961, [redacted]

5.145m The clustering of booster propellant tanks possesses an advantage over single tanking in terms of slosh stability. This is due mainly to the smaller diameters which result in higher slosh frequency and reduction of sloshing mass. The sloshing mass of the SA-1 was always less than 3% of the total vehicle mass.

5.145n Because of the current lack of ability to reliably fabricate large single tank Saturn boosters, the clustered tank configurations will probably be used for some time. The slosh suppression problems of the clustered tanks seem to be minor problems that will be engineered without undue complications arising. However, the problems associated with large diameter tanks could be much more complex and difficult to suppress, and would probably call for slosh damping stability circuits included in the closed loop control system. The weight penalty induced with extensive baffling, combined with the additional control system constraints lessen the attractiveness of large diameter propellant tanks from a control standpoint. The advantages in other technical areas of large diameter propellant tanks should be weighed against the control and cost disadvantage to evaluate whether the development effort is really feasible.

5.145o In conclusion, it may be said that slosh suppression analyses and capabilities exist for clustered tank vehicles; although the slosh analysis of the larger diameter single tanks is not more difficult, the suppression techniques may be undersirable or incapable of providing acceptable control stability.

PART III. SYSTEMS DEVELOPMENTAL OUTLOOK

Hydraulic Thrust Vector Control System

5.146 The hydraulic medium seems to be the most respected method at the present time for all stages of Saturn boost. State-of-the-art development in the cold gas cylindrical and vane actuator techniques, and the hot gas servo system is such that hardware required to implement these systems has generally lower reliability and much less operational experience than equivalent hydraulic systems. It has been stated that the gas actuator system holds merit for the second generation or follow on systems.[44] The hydraulic components of gimbal systems are prone to environmental hazards, both from engine heat and space or cryogenic cold-soaking. The working fluid of the hydraulic systems is the most temperature sensitive component in the present systems. The current design fluid specificality (Mil-0-5606) restricts the unshielded environmental extremes to -30°F and 275°F. It has been stated that acceptable all mechanical or pneumatic systems could be prototypes in 7 months.[45] This is probably very optimistic.

5.147 The RP-1 fueled 1st stage engines could use propellant fluid pressurization systems as substitutes for the hydraulic systems. Opinion, however, seems to be that the reliability gain would be negligible and the additional engineering effort expended to develop two different fluid power systems for F-1 engines would nullify what advantage that could be gained.

5.148 Both the J-2 and LR 115 engines scheduled for possible us in Saturn upper stages require gimbal capacities and hydraulic techniques comparable to systems in use (Atlas MA-3). Inputs and experience from these previous systems will probably be useful in perfecting the hydraulic components from present state-of-the-art hardware.

5.149 The single engine actuator system required for the F-1 would require 2 servo actuator assemblies (pitch and yaw) and their integrated power package. The lightest and most dependable pump power source would be an accessory drive pad. For 3750 RPM, the system would require about 114HP for periods of maximum control rates.[46] F-1 design total gimbal capacity is ± 6° in a square or ± 8.4° in the diagonal plane.[47]

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[44] General Dynamics, AE 61-0961, October, 1061, [redacted].

[45] Lockheed, ER 5388, October, 1961, [redacted].

[46] General Dynamics, AE 61-0967, October, 1961, [redacted]

[47] Ibid.

J-2 total gimbal capacity is slightly larger being $\pm 7^{\circ}$ in a square and $\pm 9.9^{\circ}$ on the diagonal. These capabilities seem to satisfy the predicted requirements.

5.150 The need for hydraulic accumulators in the system would be present unless the drive pump is sized to supply maximum need rates (4500 psi). The F-1 actuator system weight is predicted to be about 350 lbs per engine.

5.151 The J-2 and LR 115 design actuatory packages and hardware are similar but on a smaller scale (20.9 HP @ 8900 RPM).[48] Approximate actuator system weight is predicted at 80 lbs. The major components of the F-1, J-2, and LR 115 hydraulic systems are:

a. Pump.

b. Reservoir.

c. Servo actuator (piston and cylinder).

d. Relief valves and check valves.

e. Electro servo hydraulic valves.

f. Filter.

g. Fluid.

h. Auxillary pump and meter.

5.152 Both large and small systems should employ an auxillary hydraulic pump to pressurize the system at stage ignition so the main pump can be unloaded resulting in minimal turbopump starting torque. Since the hydraulic system is characterized by numerous plumbing lines and joints, development of better brazing and flex-line engine connections are applicable but not demanded in this area. Gimbal reliabilities for the 1.5×10^{6} lbs thrust F-1 engine are subject to variation from .974[49] to .9994[50] in the sources considered. Reliability indices for the two upper stage hydraulic systems were derived from the Atlas system figures and predicted to be .986 for late 1965.[51]

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[48] Ibid.

[49] Ibid.

[50] North American Aviation, Report SID 61-327, October, 1961, [redacted]

[51] General Dynamics, op. cit.

5.153 In addition to the more popular hydraulic method, there are other methods of thrust vector control. Their techniques, advantages, and shortcomings are discussed here.

Aerodynamic Surfaces

5.154 Aerodynamic fins cause performance penalties associated with high drag and does not provide adequate control either in the early or late portion of the trajectory where dynamic pressures are negligible.

Fluid Injection

5.155 Fluid injection techniques are becoming more feasible and showing results with solid engines. The method involves injection of a high pressure gas into the divergent nozzle section, resulting in an oblique shock pattern which effectively deflects the thrust vector up to 6°.[52] Advantages are the use of non-movable nozzles minimizing the hydraulic or mechanical problems associated with gimbal techniques.

5.156 Development state-of-the-art is still early, primarily with solid motors, and is characterized by complex plumbing, injection, and proportional valving hardware. It is an interesting technique and would be well suited for space requirements as it is relatively insensitive to cryogenic environments that affect hydraulic systems. It could prove to be a highly productive technique.

Jetavators and Jet Vanes

5.157 These techniques have proven very reliable on smaller solid boosters such as Seargent, Pershing, and Scout. But for engines on the current scale they tend to be heavy and unwieldy. They are subject to erosion and would not provide effective control toward the later stage of trajectories of the Saturn size vehicles.

Rotatable Solid Attitude Motors

5.158 This technique, along with the secondary fluid injection methods seem to be the best choices for development as follow-on or second generation thrust vector control systems. Allison Corp. has done work on developing a solid motor with a ninety degree deflected nozzle. The motors rotate as a unit and control forces are provided by rotating them singly or in sets. In the neutral position, the nozzle is oriented to the rear and aligned with the main thrust. The motor burns throughout launch until staging, oriented in the neutral position until control is required then rotated by electric or hydraulic means so the sideward thrust provides the control adjustment.

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[52] North American Aviation, op. cit.

5.159 State-of-the-art development has disclosed inherent system problems such as: large power requirements for rotating motors of large thrust (100,000 lbs class), frequency response of large systems, development of 90° nozzle of sufficient life, thrust termination, and control system development. The system is considered feasible for large first stages and seems to have less serious developmental problems than other new thrust vector control systems such as secondary fluid injection.

5.160 The significant advantage is that it also allows the use of fixed nozzles on the main booster and the thrust vector control system. Also, during development phases, the firing of the main motor does not have to be undertaken to test the development of the thrust vector control system.

5.161 Development is in the early stages and seems to be oriented toward solid boosters. This technique is probably not going to replace the first stage liquid engine hydraulic system unless some unexpected usefulness is attained; but it also could prove very feasible as an attitude control system for periods of thrust (liquid or solid engine) in space after long periods of exposure to space environments.

5.162 If the problem of protecting a hydraulic system against low temperatures for long periods proves insurmountable, the follow-on systems that appear most promising at the current time are the pneumatic or mechanical systems followed by the rotatable solid grain and the secondary injection technique.

Flight Control Computer Placement

5.163 The typical signal generation methods for the control methods above would be analog systems that employ various combinations of amplifiers, modulators, and demodulators to convert the guidance commands into meaningful current to drive the servomechanisms that regulate the thrust control actuators. Digital systems are becoming operational but have not been exploited to advantage in the large booster control area. In a multi-stage vehicle such as the Saturn type Apollo boosters, there are two choices for setting the requirements for flight control units electronic capacities. (1) To place the control units for each stage in the applicable stage and provide only a minimal control system with the payload; or (2) place all the stage control electronics in a more complex integrated control computer with the payload. The vehicle control compensation requirements are usually satisfied by rate gyros necessary in the individual stages but which would provide signals to the upper stage computer only.

5.164 The calculated reliabilities of the two choices seem to be equally acceptable. The use of the individual flight control units in each stage is currently utilized in the Minuteman program. The choice involves such parameters as wiring elimination, less switching and phase matching, and each actuator receiving a more valid signal. The concept has demonstrated a high level of success in the Minuteman vehicle. There is room for study in the Saturn-Apollo program of these choices in terms of such variables as complexity, adaptability, weight, power, checkout integrity, control and development, and reliability.

5.165 The basic concepts of thrust vector control for the current vehicles intended for the Manned Lunar Program are attainable in terms of existing hardware or adaptations of the methods discussed above. The reliabilities or man ratings consistent for manned flight are also available. The problems are generally of an engineering nature involving the fitting of state-of-the-art hardware and methods to the specific vehicle parameters and mission profiles when they become known or defined by NASA.

<u>Auxiliary Control Rocket Systems</u>

5.166 Besides the main propulsion cluster, there are at least two other auxiliary propulsion systems included on the Apollo space capsule and various mission modules: (1) the very small thrust altitude reaction jet system, and (2) the larger thrust vernier engine system used for mid-course corrections and terminal orbital maneuvers around the moon.

5.167 These auxiliary rocket engines and their development problems really lie within the technical area of propulsion systems; but the results of these systems are so integrated with the control functions and systems that the major points in their developmental outlooks will be discussed here.

<u>Altitude Jet Systems</u>

5.168 There are several methods available to influence the altitude of a capsule or satellite in free-fall space. However, these other techniques will be discussed little, if at all, since there are generally not applicable to the size vehicles and missions present in the Manned Lunar Mission. Some other techniques are:[53/]

a. Solid mass expulsion (bullets, etc.).

b. Plasma reaction jet.

c. Single axis and spherical flywheels.

d. Solar radiation control vanes.

e. Gravity dipole shape.

53/ Walter Haeusserman, <u>Comparison of Some Actuation Methods for Altitude Control of Space Vehicles</u>, IAS Manned Space Station Symposium, April, 1960.

5.169 Briefly, it may be shown in the comparison of the electric plasma rocket and the chemical rocket that the savings of propellant weight due to the high specific impulse of the electric rocket is partly offset through the heavy equipment which must be carried to supply the electrical power. Only over an extended period of thrust will the propellant mass of the chemical rocket exceed the weight of the auxiliary power supply for the electrical rocket. In addition, the development of chemical propulsion has provided considerably more respect, through age, for the chemical systems.

5.170 Initial design parameters[54] provide that the space capsule and modules be equipped with a reaction jet system using storable hyergolic fuels as propellant. The choice of bipropellant hypergolic fuels will require some further developmental efforts. But, the required solutions probably are within reach and the advantages in terms of performance, storability, and handling and ignition characteristics will undoubtedly offset the requirement for advanced technology. Other choices of fuels could have included solid propellants, which possess undesirable on-off throttling over the long term; or monopropellants, which are well suited to the requirements but require catalyst structures within the chamber, and sometimes possess undesirable handling and cold start characteristics.

5.171 As design refinements are incorporated, it may be possible to change to the main propellants, H_2 and O_2, stored as gas in small tanks. In fact, H_2 is a decent propellant by itself and gives a specific impulse of 200 seconds if heated to $270^{\circ}R$.

5.172 For effective altitude control, the thrust requirements usually are under 25 lbs. These low thrust levels are attainable and working design hypergolic systems are reported down to as low as approximately 1 lb.[55]

5.173 The actual control of the altitude jets will be through pulse-counting techniques and not variable throttling. This scheme will result in at least two complete altitude control systems, one of high thrust (30-40 lbs) and one of low thrust (6-8 lbs) for fine adjustments. Currently, the fixed injector valving simplicity is more desirable than the design of a proportional injector valve with variable flow since difficulty increases as thrust level decreases.

54/ Project Apollo Spacecraft Development, Statement of Work—Phase A, NASA Space Task Group, July, 1961, [redacted]

55/ Vickers, Inc., Research and Development Laboratories.

5.174 The primary objective of the pulse rocket is to produce a very short time-width pulse of high repeatability and reliability. Specific impulse is a measure of efficiency and reflects in the performance-weight ratio. The desired time width pulses are on the order of milliseconds and output would ideally be of square wave configuration. These requirements mean very low ignition delay times and propellant valving located at or very near the injector.

5.175 With the perfection of short time repeatable pulses, total impulse control is possible by simply calculating the required number of pulses and counting them at maximum frequency to the propellant inlet valve. An additional feature is that moderate thrust level adjustment may be attained through frequency control of the pulsing inlet valve. Very small pulse units are ideal for minimum limit cycle operation as the smaller and more accurate the pulse, the narrower are the resultant deadband limits of the closed loop control system.

5.176 At the current time, laboratory rockets are producing preliminary results which are very satisfactory. Pulse widths of 10 milliseconds of generally square wave shape, thrust buildup of about 1 millisecond preceded by 7 millisecond valve operation times are attainable with a 25 lb thrust rocket using hypergolic propellants and commercial solenoid components.[56/] Standard deviations or uncertainties in pulse width and delay times are not in evidence and it would be assumed that they are undesirably high since the techniques are still in preliminary stages.

5.177 One of the major areas of concern in smaller rocket engines is the cooling problem; for engines below 100 lbs thrust the regenerative cooling technique is of questionable use since the coolant passages become quite small and restructive. Local wall overheating from nonuniform coolant flow is a hazard. The filling and purging of the coolant passages is a major factor why regenerative systems cannot be used on pulse reaction control since the effect on system response is disastrous.

5.178 In hypergolic engines, using hydrazine as coolant, and having a shutoff system to reduce the starting and shutdown transients gives rise to the problem of dissipating the residual heat in the chamber materials without violent decomposition of the hydrazine trapped in the coolant passages. There are some suggestions[57/] to solve these problems, but they are complicated and would not really justify the effort involve

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[56/] Conners and Latto, Characteristics of Small Control Rockets, NASA.

[57/] Ibid.

5.179 Much better return on effort is available and demonstrable in small rocket cooling in the areas of ablation and radiation techniques. Very small thrust engines (0.5 to 1.2 lb) do not have heat transfer problems and have been tested[58] for prolonged periods without throat erosion which is extremely undesirable in altitude control, since throat erosion is related to thrust variation. A problem area could easily develop in small rocket cooling if rockets on the order of 50 to 100 lbs thrust are deemed necessary for space vehicle altitude or vernier engine use for time periods that would produce heat transfer problems.

5.180 In retrospect, the area of small hypergolic liquid rocket systems needs additional effort in order to meet the mission requirements, but preliminary laboratory results seem to point out that solutions are within reasonable reach and will be available within a year or two.

Vernier Velocity Engines

5.181 The larger vernier engines which are to be used for midcourse manevers and technical lunar maneuvers are also stated in the previous reference[59] to be hypergolic engines but if much larger size (approximately 3000 fps capability). Engines of this size should be no particular problem, but this is entirely a propulsion problem and will be left to that technical area. The vernier engines will have gimbal systems which will provide thrust vector control for midcourse maneuvers, lunar maneuvers, and the lunar take off. These hydraulic problems are discussed in the previous discussions and the section on flight disturbance control.

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[58] R and D Labs, Vickers, Inc.

[59] NASA Statement of Task for Apollo—Phase A.

CONTROL PROJECT C-1

FINE GRAIN WIND STRUCTURE DETERMINATION

1. Task Statement. Improve techniques for quick evaluation of fine grain wind structure to altitudes of 60,000 ft.

2. Justification. Wind moments are the source of greatest loading on the booster vehicle. Wind gust data and vehicle stability are prime inputs to the choice of gain rates and deflection capabilities of thrust vector control systems. Methods of insuring that the fine grain gusts are actually below predicted velocities just prior to launch are required along with wind data to further adapt control systems to actual conditions.

3. Present Status. Present radiosonde balloon predictions are based on fairly rough grain wind envelope structures. Smoke trail rocket systems are showing improvements in solving this problem for altitudes of 1000 to 50,000 ft, but data reduction of information is too slow. Possible Air Force radar—high pressure ballon techniques hold some merit in this area.

4. Criticality. Methods should be refined before flight tests on the large Saturn boosters get underway on a large scale.

5. Applicability. Methods of this type will also provide volumes of data input to allow follow up control design to cope more effectively with large variances in wind gust and the resultant disturbances on large boost vehicles:

   Earth Launch and Orbit Mission.

6. Reference. Analysis of Control, paragraphs 5.128-5.133.

CONTROL PROJECT C-2

BOOSTER CONTROL REQUIREMENTS FOR WINGED PAYLOADS

1. Task Statement. Determine the control system parameter dealing with the techniques to stabilize and control a booster carrying a winged payload.

2. Justification. Studies have indicated that Saturn boosters will be capable of launching earth orbital or translunar winged payloads.

3. Present Status. Current work does not evidently consider stability analysis and wind envelope effects associated with winged payloads as inputs to Saturn control system design.

4. Criticality. To be most effective, this information should be generated before Saturn systems are frozen, but it depends on the decision to use winged payloads with Saturn vehicles.

5. Applicability. To any winged payload program; all missions.

6. Reference. Analysis of Control, paragraphs 5.130-5.133, 5.102, 5.105.

[redacted]

CONTROL PROJECT C-3

ULLAGE CONTROL TIME PARAMETERS FOR LIQUID ENGINE STARTABILITY

1. Task Statement. Determine time frames and ullage forces required for engine startability under zero g; and determine the maximum zero g time frame under which an engine may be started without external ullage control necessary.

2. Justification. Because of floating fuel at zero g, engine startability is affected. Longer periods of ullage thrust are capable of perturbing a flight path to a small extent. Unnecessary ullage control at staging sequences incurs reliability and weight penalties at liftoff.

3. Present Status. No evidence of work concerning maximum coasting time without ullage control at staging. Work available on g level required but no evidence of correlation with time frame of application required or effects of vibration or spin on ullage control for engine start.

4. Criticality. The values and relationships are needed to provide a maximum reliability for orbital starting but not to incur weight penalties for more ullage control systems than are necessary.

5. Applicability. To all staging, orbital launchings and midcourse corrections utilizing restartable liquid engines.

6. Reference. Analysis of Control, paragraphs 5.25-5.28, 5.45, 5.46, 5.136, and 5.143.

CONTROL PROJECT C-4

ATTITUDE JET RELIABILITY AND REDUNDANCY STUDY

1. Task Statement. Examine thrust ratios, redundancy requirements and capabilities of attitude control system to reduce present system weight and provide guidance for follow on variable thrust systems.

2. Justification. Hypergolic variable thrust systems are in basic development. Present attitude jet systems are over redundant and have moderately high but questionable reliability.

3. Present Status. H_2O_2 system redundancy on the Mercury capsule provide for 18 attitude nozzles. System manufacturer gives reliabilities as 88.7% and 90.48% for the automatic and manual systems respectively.

4. Criticality. The work should be completed in time to be used in development of follow on systems for Gemini or Apollo earth orbital testing prior to lunar landing sequences.

5. Applicability. Extremely high applicability due to the requirement for attitude control in all manned flights and in an increasing number of probes.

6. Reference. Analysis of Control, 5.36, 5.39, 5.41, 5.170-5.180.

CONTROL PROJECT C-5

ENGINE THRUST UPRATING TECHNIQUES FOR ENGINE OUT CONTROL VEHICLES

1. Task Statement. Learn to uprate single engine thrust in order to replace lost thrust from an engine-out control command.

2. Justification. Engine-out systems are of questionable mission value unless engine output can be uprated, payload reduced upon loss of an engine in the cluster, or the mission can be accomplished in the reduced thrust mode.

3. Present Status. Engine-out systems are operational. Thrust uprating techniques have not developed since engines are usually operating near or at their limits for present missions.

4. Criticality. Would be highly desirable in second stage engines such as the M-1 which is still in design phase.

5. Applicability. To all liquid engine boosters designed for engine out systems, applicable to all missions.

6. Reference. Analysis of Control, 5.106-5.111.

CONTROL PROJECT C-6

ENVIRONMENTAL PROTECTION FOR HYDRAULIC CONTROL SYSTEMS

1. Task Statement. Study environmental protection for hydraulic control system components to insure initial reliability after long periods of spacial temperature and vacuum environments.

2. Justification. Hydraulic systems are required to operate accurately and reliably in the lunar flight plan after 12 to 175 hours spacial exposure.

3. Present Status. Present designs consider electrical or insulation protection; both are either unreliable or demand too much electrical power.

4. Criticality. Must be available for orbital rendezvous test flights.

5. Applicability. If economical protection procedures are available, the solutions would be applicable to all orbital or spacial controlled missions. If environmental problems are insurmountable, results would provide input for follow on pneumatic or mechanical system design.

5. Reference. Analysis of Control, 5.63-5.67, 5.123-5.125, 5.146, 5.162.

CONTROL PROJECT C-7

EARLY BOOST ROLL CONTROL STUDY

1. Task Statement. Study roll control servoamplifier low gain limit cycle for Saturn vehicles with roll control moment coefficients from 12 to 70.

2. Justifcation. High roll control moment coefficients tend to produce roll overcontrol at initial boost.

3. Present Status. Saturn SA-1 launch evidently had no problem but it was a very early flight test launch. Atlas configurations have problems in this area.

4. Criticality. Early roll orientation is of essential value in determining accuracy of program into orbit injection.

5. Applicability. To all vehicles requiring roll control in ascent:

   Earth Launch and Orbit Mission

6. Reference. Analysis of Control, 5.23 and 5.24.

CONTROL PROJECT C-8

ATTITUDE DRIFT CORRECTIONAL IMPULSE EVALUATION

1. Task Statement. Evaluate the feasibility and philosophy between incremental and semi-continuous attitude drift control for different orbital transfer missions so that some basic tracking, guidance, and control system parameters may be evaluated for minimum weight to meet mission requirements.

2. Justification. Orbital transfer mission vehicles are prone to large inherent attitude drift while in unpowered sequences. In some instances, almost continuous attitude control is more feasible than large incremental control. Other systems could depend on continuous attitude control to replace some of their servo capabilities thereby reducing weight and power requirements.

3. Present Status. Present thought favors incremental control for most missions. This is probably justified although no orbital transfer shots have been fired yet. Any orientation system will satisfy the needs of orbital transfer whether it is the best solution in terms of weight, fuel, and system interaction is the unanswered problem at hand.

4. Criticality. All early lunar shots using rendezvous techniques will use orbital transfer sequences. These early shots will also have the most rigid weight constraint.

5. Applicability. To all orbital transfer operations, and could be extended to space flight drift control.

6. Reference. Analysis of Control, 5.30 and 5.32.

CONTROL PROJECT C-9

NON-SPIN ATTITUDE CONTROL REQUIREMENTS PRIOR TO SPACIAL RESTARTS

1. Task Statement. Determine the degree of angular orientation accuracy required of the attitude control system so that upon restart the gimballed vector control components can make final closed loop adjustments with a minimum of thrust deflection and structural bending interference.

2. Justification. Due to the many restarts resulting from orbital transfer orbital launch, and midcourse corrections, there could be unnecessary fuel and weight penalties associated with requirements which are too strict or too broad in relation to the initial attitude of the vehicle. Bending mode disturbance will be present in orbital launch and early midcourse corrections; these should be considered in setting the desirable levels and rates of deflection on the gimballed engines at ignition.

3. Present Status. Current investigations do not isolate this area of study in terms of the required mission or the number of restarts required for the mission. Bending modes of orbital launch vehicles joined at the midsection in orbit, have not been analyzed for adverse responses to thrust vector control rates and angles upon orbital ignition.

4. Criticality. Knowledge of the above values should be ascertained before orbital transfer, and rendezvous or docking mission are undertaken since each is characterized by spacial restarts.

5. Applicability. All phases of lunar or planetary exploration, both manned and unmanned, are dependent upon midcourse restarts for trajectory corrections.

6. Reference. Analysis of Control, 5.29, 5.44, 5.47 and 5.48.

CONTROL PROJECT C-10

UPRATING CONTROLLABLE VARIABLE THRUST RATIOS IN LIQUID FUELED ENGINES

1. Task Statement. Uprate controlled thrust ratios of 50:1 to 100:1 range for liquid engines of moderate (100,000 lb) thrust level, utilizing liquid propellants that will store for at least 60 hours to enhance successful lunar manned rocket landing.

2. Justification. Present fixed thrust retro systems are unable to provide reliability and reproducibility of thrust to provide safe lunar manned landing. Present variable thrust engines have low thrust ratios for a single stage automatic lunar landing.

3. Present Status. Present development has yielded thrust ratios of up to 25:1. Use of manned control systems may allow successful landings with less than 50:1 thrust ratio. Two stage landing systems will reduce required thrust ratios; and in turn reduce reliability.

4. Criticality. Should be developed for automatic soft unmanned lunar landings and subsequent manned landing.

5. Applicability. Manned and unmanned soft lunar landings, possible extension into velocity midcourse control.

6. Reference. Analysis of Control, 5.71-5.75 and 5.77.

CONTROL PROJECT C-11

CONTROL SYSTEM ASPECTS OF SLOSH SUPPRESSION IN LARGE DIAMETER PROPELLANT TANKS

1. Task Statement. To determine the feasibility or need for dependence upon the control computer for slosh damping response through structural loading feedback signals.

2. Justification. Liquid propellant systems are prone to liquid slosh instability during ascent motion. Small diameter tanks rely on internal damping devices to provide slosh suppression. Large diameter tank suppression devices may not provide acceptable action or may incur weight penalties which would cause a shift of technique to control system feedback stability similar to structural bending stability control.

3. Present Status. Internal devices are capable of acceptable suppression without major control system interaction for tank diameters up to 120 inches. Saturn single tank models would probably range to 350 or 400 inches in diameter if fabrication methods became more capable. FY 61 and 62 funding includes many fabrication and materials studies leading toward better large tank fabrication methods.

4. Criticality. The outputs of this study should be examined to provide guidance as soon as possible toward continuance or cancellation of expenditures to fabricate large tanks; and inclusion, if required, of sloshing parameters into control systems now in development to be ready for large-tank vehicles.

5. Applicability. Earth launch and orbit missions for large liquid boosters. Possible extension into the orbital launch missions for liquid propellant vehicles.

6. Reference. Analysis of Control, paragraphs 5.145j-5.145o.

[redacted]

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF CONTROL

CATEGORY: MISSION ORIENTED CONTROL STUDIES

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Orbital Launch Guidance and Control Studies	FY62/$50K	G&C	Methods are applicable, scope is questionable.
2.	Analytical Determination of Rigid Body Motion of an Orbiting Body	FY61/$50K FY62/$50K	RP RP	Appears unwarranted, or of relatively little value.
3.	Hot Gas Reaction Nozzle System	FY61/$81K	G&C	Useful, but probable duplication.
4.	Mechanical and Hot Gas Actuators to Replace Hydraulic Actuators	FY61/$70K	G&C	Should consider whether hydraulic environmental protection problems are insurmountable before replacing the hydraulic systems.

General Remarks: #4 should consider additional problems of space environment such as lubrication or space cold welding. #1 is very useful approach, but includes some funding for Venus trajectory launch and correction studies which are not directly applicable to the manned lunar mission.

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF CONTROL

CATEGORY: DISTURBANCE CONTROL STUDIES

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Basic Empirical Wind Shear Investigation	FY61/$190K	AERO	Should be preceded by attempts to cut data processing time on current smoke trail techniques. Useful, but possible duplication with Air Force.
2.	Transister Active Filter for Servo Loop	FY61/$100K	G&C	No particular criticality or need.
3.	Digital Filters for Control Systems	FY62/$ 75K	G&C	Applicable. Undertake.
4.	General Study of Interactions of Motion of Liquids in Containers and Vibratory Behavior of Load Carrying Structures under Actuation from External Sources of Time Dependent Magnitude	FY62/$120K	S&M	General approach is of questionable effort since slosh dynamics are extremely dependent on vehicle characteristics. Present clustered tank slosh suppression is state-of-the-art.
5.	Theoretical Research on Non-Spinning Spacecraft (Multi-stage) performing Bending Oscillations	FY62/$ 50K	AERO	
6.	Stability and Control. Load Distributions	FY62/$100K	AERO	Should include buffeting from escape tower shapes.

CATEGORY: DISTURBANCE CONTROL STUDIES (Cont'd)

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
7.	Environmental Criteria Design Study	FY62/$ 65K	AERO	Of little use until more accurate fine grain wind shear data gathering methods are available as partial inputs.
8.	Analytical Synthesis for Transfer Functions	FY62/$ 50K	AERO	Probable duplication.

EVALUATION OF TECHNOLOGY SUPPORT PROGRAM FOR TECHNICAL AREA OF CONTROL

CATEGORY: GENERAL CONTROL STUDIES

	Project Title	Funding	Origin (MSFC Laboratory)	Remarks
1.	Solid State Gain Changers for Control Systems	FY62/$ 50K FY63/$100K	G&C	
2.	Radiation Effects on G&C Equipment Aboard Nuclear Powered Vehicles	FY62/$ 75K	G&C	No control systems proposed yet for nuclear engines. Also probable duplication of effort.
3.	Adaptive Control Studies	FY61/$200K	AERO	
4.	Seals and Sealants for Space Applications	FY61/$ 25K	ORDABDSN	Should include effects of hydraulic synthetic fluids on seals for long periods.

General Remarks: #2 is of questionable use at the present time and should be re-evaluated. #1 and #3 should include the entire trajectory of the Apollo landing mission if they do not already.

APPENDIX A

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APPENDIX B. MISSION DEFINITIONS

INTRODUCTION

B.1 The ultimate primary mission of the manned lunar mission is the safe passage of a manned vehicle to and from a lunar landing. However, prior to accomplishment of this mission, there are other primary missions to be accomplished falling into the cislunar, circumlunar, lunar orbit, and finally lunar land categories. Each mission is a sequence of events that have been grouped together to form secondary missions.

B.2 Table B.1, is a Primary Mission/Secondary Mission/Event Matrix which also shows the difference between the mission event sequencing of direct and interrupted flights.

B.3 This Appendix defines the secondary missions, the basic units with which the technical areas of the manned lunar mission were analyzed.

<u>Earth Launch and Orbit</u>

B.4 This mission is defined as the transfer of a payload from the earth surface into a low altitude orbit. The need for first placing the payload on this low altitude "parking" orbit is justified by the following considerations:

a. Many of the propulsion systems to be made available within the time frame 1960-1975 will lack the capability of placing a manned vehicle into a direct lunar trajectory. The vehicle must first be assembled and/or refueled in space. The resulting rendezvous

TABLE B.1

PRIMARY MISSION/SECONDARY MISSION/EVENT

EVENTS	SECONDARY MISSIONS	PRIMARY MISSIONS: Cislunar Flight		Circumlunar Flight		Lunar Orbit		Lunar Land	
		Direct	Interrupted	Direct	Interrupted	Direct	Interrupted	Direct	Interrupted
Launch Pad Checkout	Earth Launch & Orbit	X	X	X	X	X	X	X	X
Earth Launch		X	X	X	X	X	X	X	X
Earth Ascent		X	X	X	X	X	X	X	X
Parking Orbit Injection			X		X		X		X
Parking Orbit Sustenance			X		X		X		X
Orbit Transfer			X		X		X		X
Earth Orbit Injection			X		X		X		X
Earth Orbit Sustenance			X		X		X		X
Outbound Orbital Rendezvous	Orbital Rendezvous		X		X		X		X
Outbound Orbital Docking	Orbital Docking		X		X		X		X
Post Dock Checkout			X		X		X		X
Material Pre-Transfer Checkout	Orbital Transfer, Assembly, Repair, Maintenance & Checkout.		X		X		X		X
Material Transfer			X		X		X		X
Pre-Orbital Launch Checkout			X		X		X		X
Earth Orbital Launch			X		X		X		X
Outbound Translunar Trajectory Injection	Earth Orbital Launch & Translunar Flight	X	X	X	X	X	X	X	X
Outbound Translunar Trajectory Sustenance		X	X	X	X	X	X	X	X
Outbound Midcourse Correction		X	X	X	X	X	X	X	X
Outbound System Checkout		X	X	X	X	X	X	X	X
Lunar Orbit Injection	Lunar Landing.					X	X	X	X
Lunar Orbit Sustenance						X	X	X	X
Lunar Descent								X	X
Lunar Hover or Land								X	X
Lunar Pre-Launch Checkout	Lunar Launch & Transearth Flight							X	X
Lunar Launch								X	X
Lunar Ascent								X	X
Lunar Orbit Injection									X
Lunar Orbit Sustenance									X
Lunar Orbital Launch						X	X		X
Transearth Trajectory Injection			X		X	X	X	X	X
Transearth Trajectory Sustenance		X	X	X	X	X	X	X	X
Transearth Midcourse Corrections		X	X	X	X	X	X	X	X
Transearth System Checkout		X	X	X	X	X	X	X	X
Earth Orbit Injection	Earth Re-entry & Land		X		X		X		X
Earth Orbit Sustenance			X		X		X		X
Inbound Rendezvous									X
Inbound Docking									X
Earth Re-entry		X	X	X	X	X	X	X	X
Earth Land		X	X	X	X	X	X	X	X

operations, to be discussed later, are best accomplished when the mating parts or vehicles are first placed in the parking orbit.

b. Favorable conditions for launching a vehicle possessing adequate propulsion systems into a direct earth-moon trajectory occur very rarely. Launching into a parking orbit offers almost a daily opportunity of initiating a lunar flight or of effecting orbital rendezvous according to a predetermined sequence.

B.5 Successful launching and placement of a payload into a parking orbit is contingent upon the solution of a number of problems. This investigation will discuss only these problems associated with the operation of the systems intrinsic to the launch vehicle during the duration of the mission. These problem areas involve the structure of the vehicle, the operation of its guidance and control systems, communications between vehicle and earth stations and so on. Excluded from consideration in this study are (a) problems which may arise before the initiation of the mission, that is, before launching (prelaunch planning, check out, countdown, etc.); (b) problems involving exclusively earth based systems (communication between earth tracking centers, earth based data processing operation, etc.); (c) problems directly associated with the design, operation and reliability of the vehicle propulsion systems, except insofar as the effects of the environment generated by these propulsion system (aerodynamical, acoustical or thermal stresses) on the components of the vehicle.

B.6 On the basis of the preceding considerations, the launch mission may be considered to begin on the launching pad. The limited number of launching pads now available at Atlantic Missile Range may prove to introduce serious problems in fulfilling missions requiring a rigid schedule of launching, for instance, the assembly in space of many components or the sustained resupply of a large lunar base. The location of launching facilities also generate problems affecting the mission requirement. For example, the inclination of the lunar orbital plane will be approximately equal to the latitude of AMR on or about 1969. At that date, opportunities for launching vehicles within the lunar orbital plane will occur daily at Cape Canaveral. The situation will deteriorate during subsequent years until 1978, when launchings from AMR will require, at least a 10° dogleg maneuver to bring the vehicle within the lunar orbital plane. The mission will thus be penalized by increased guidance errors and fuel requirements. Ideal conditions would be restored by having launch facilities at the proper latitude, say Puerto Rico. Considerations of the effect of location and availability of launch facilities, however, cannot be discussed in detail at this stage. It will be assumed that ideally located launch sites are available as needed by the NASA mission.

B.7 The launch facilities requirements vary greatly with the frequency of the launchings dictated by the mission. It is anticipated that NASA ultimate objectives, that is, the type of manned lunar mission can be defined at some future date in order to make it possible to integrate these problems into the general investigation.

B.8 Termination of the launch mission is defined as the moment when the earth launched vehicle is injected into a lunar trajectory or into a Hohmann transfer orbit for subsequent rendezvous with the orbiting platform. The problems associated with these tasks are discussed in a subsequent section.

B.9 The launch-orbiting mission is subject to a number of basic requirements imposed by the nature of the lunar vehicle. These requirements involve primarily (a) the altitude of the parking orbit as well as the accuracy with which this orbit must be achieved and (b) the payload to be placed into the parking orbit.

a. It is shown, in the discussion of the orbital rendezvous mission, that the altitude of the earth orbiting station must be in the neighborhood of 300 nautical miles to reduce aerodynamic drag and minimize the radiation from the Van Allen belts. In order to reduce the hold off time on the parking orbit, the altitude of the latter must be as low as possible, consistent with a reasonable vehicle life. A parking altitude of 100 nautical miles is generally considered as the best compromise between these opposite requirements.

b. The payload of the lunar vehicle varies from an absolute minimum of 15000 lbs to several hundred thousand pounds, depending upon the nature of the manned capsule and the mission requirements. The minimum refers to the weight of the Apollo capsule, injected into a ballistic circumlunar trajectory with ballistic reentry into the earth atmosphere. This payload does not provide for midcourse maneuvers, almost certainly required for placing the vehicle into the proper circumlunar entry/exit corridor; to insure ballistic earth capture on the return leg as well as restricting the landing area to locations where the capsule can be reliably recovered. The fuel requirements for these midcourse maneuvers may be slightly less if the manned capsule has lift capabilities (winged payload).

B.10 The payload is increased to approximately 125,000 lbs for an Apollo capsule with lunar landing and takeoff capabilities, and to still higher value if midcourse maneuvers have to be provided for. The payload, of course, determines the severity of the problems associated with the launch missions.

Orbital Rendezvous

B.11 This mission consists of bringing a vehicle (chaser) in close proximity to an earth orbiting body (target) for the purpose of transferring fuel or crew from one vehicle to the other, assembling a more complex structure, inspection of a vehicle, and so forth. The rendezvous mission will concern either an earth launched vehicle (outbound rendezvous) or a vehicle on the return leg of the lunar trip (inbound rendezvous).

B.12 The rendezvous mission is considered to be initiated at the time the chaser acquires the target, at which time a sequence of tracking, data processing and guidance operations is initiated. The mission terminates when chaser and target are in such close proximity that the guidance systems cease to operate and inertial or mechanical docking maneuvers are initiated. In rendezvous missions it is assumed that the major maneuvering capabilities are restricted to the chaser; control and other minor maneuvering can be conducted by both vehicles.

B.13 Three modes for directing the chaser to the rendezvous are conceivable:

a. The tracking and guidance functions are performed by the orbital element (target) and decisions communicated to the maneuverable chaser.

b. These functions are performed by the chaser itself.

c. The primary guidance is performed by ground based stations. The command decisions are communicated to the chaser.

Each of these modes introduces various tracking, data processing and communication problems, whose impact may be expressed in terms of cost/accuracy/reliability trade-offs. Only when the ultimate objectives of the manned lunar mission have been defined will it be possible to assign a relative value to trade-off constituents such as cost and reliability, and will it be possible to select the most desirable technique.

B. 14 Outbound rendezvous missions are an integral constituent of a manned lunar mission based on Saturn launch vehicles, which have no capabilities for direct lunar flight. Inbound rendezvous missions have a more remote (second generation vehicle) applicability since the manned Apollo capsule will probably be designed for ballistic reentry into the earth atmosphere. However, the study of inbound rendezvous missions cannot be summarily set aside until it is conclusively proved that the manned capsule can be safely guided into an earth capture orbit and reliably recovered by ground support activities.

B. 15 Basically the problems raised by inbound and outbound rendezvous missions are similar. However, the problems of target acquisition, tracking and guidance for inbound missions may prove to be the more severe, unless the earthbound vehicle can be directed into the orbital plane of the target and unless the velocity vectors of the two vehicles can be adequately matched. These requirements are considered in the section devoted to the lunar launch missions.

B. 16 Orbital rendezvous missions are subject to a number of basic requirements imposed by space flight mechanics, by the characteristics of the vehicles and by the nature of the lunar mission. A few of these requirements are considered in the following:

a. The altitude of the earth orbiting target is limited toward high values by the atmospheric drag which in turn decreases the accuracy with which the orbit ephemeris can be determined. Higher altitudes also increase the detection range of ground tracking stations. However, the altitude is limited by the effects of Van Allen radiations on the target, vehicle and crew. The accepted rendezvous altitude appears to lie in the neighborhood of 300 nautical miles.

b. The optimum trajectory between parking and target orbit in outbound rendezvous on the basis of fuel economy, is the Hohmann transfer orbit. This transfer, however, can be initiated only when the two vehicles are in the proper positions in space. The waiting period for such condition to occur increases the energy requirements for tracking or communications and generally increases the probability of failure of the mission, all the more so when several rendezvous must be achieved before orbital launch. In inbound rendezvous missions the problems are compounded by the potential effects of Van Allen radiations on the manned capsule if the latter is parked on an orbit of higher altitude than that of the target. Analysis of rendezvous missions should thus properly include consideration of the trade-off between increased fuel requirements for non-Hohmann transfer and penalties resulting from the degradation in mission reliability.

c. The reliability of the entire lunar mission decreases as the number of rendezvous required to assemble or refuel the lunar vehicle increases. This analysis must include a study of means for minimizing the effect of a rendezvous failure on the entire operation.

Orbital Docking

B.17 Orbital docking is the mating of two vehicles in space to form an assembly that subsequently acts as a single vehicle. The docking technique may consist of the mating of major assemblies of vehicles, or it may be nothing more than the interconnection of two vehicles by a cable, device or a subassembly. However, there must be no relative motion of one vehicle with respect to the other. Chronologically, orbital docking begins when terminal guidance in rendezvous ends; it ends with the mating of the two vehicles under consideration.

B.18 Orbital docking is a mandatory prerequisite to fuel, man or equipment transfer between orbiting space vehicles. It is a mission that must be proven to be successful before more sophisticated orbital missions can be undertaken.

B.19 If the lunar mission is to be undertaken by direct flight, orbital docking need not be accomplished. However, it is very possible that an indirect or interrupted route will be utilized in which the launch vehicle will require refueling at some orbital station. Thus, orbital docking could be an extremely important maneuver within the lunar mission.

B.20 During the docking operation, the main propulsion engines of vehicles will be off. The vernier and attitude control engines will be operational; the status of these engines will be dependent on the final adjustment of vehicle velocities and final orbital alignment between vehicles necessary for docking. When docking begins, the relative velocity between the two vehicles should be at a minimum. Terminal guidance should have aligned the vehicles such that orbital misalignment in the radial direction, and axial angular misalignment is insignificant. Mitigation equipment will decrease contact shock to either of the vehicles to an insignificant value. The torques resulting from angular misalignment of mating vehicles should not affect either vehicle or the combined assembly in pitch, roll, or yaw. The docking operation is a comparatively short orbital maneuver; docking time should be in the order of a few minutes.

B.21 During final phases of docking, the vehicles will probably be too close for radar tracking to be accurate; therefore visual (television) or photoelectric tracking and control for final alignment will probably be utilized. Communication between vehicles will be telemetered; audio-radio will also be used if both vehicles are manned. Although the docking procedures will probably be automated, the crew should be able to dock manually and control final docking procedures. As the vehicles come together, the manned vehicle, whether it be the target or chaser, should be able to control the unmanned vehicle. The distance between vehicles, the closing rate, and the orbital angular and axial misalignments will be monitored as often as possible; this information will be analyzed by the guidance system so that final control can be incorporated and/or the mission could be aborted if dangerous conditions exist. If both vehicles are unmanned, these data will be telemetered to the monitoring ground station for subsequent commands.

B.22 Following the coupling of the vehicle, the vehicles should have automatic checkout to assure that connections at the interfaces between the two vehicles have been made properly. It should be possible to check out both manned and unmanned vehicles from the manned vehicle. If both vehicles are unmanned, the checkout data will be telemetered to ground stations.

B.23 Orbital docking will probably occur at altitudes approaching 300 nautical miles above the surface of the earth. The docking missions should be reliable in spite of the environments in which docking will occur and the vehicles will have undergone prior to docking.

Orbital Transfer, Assembly, Repair, Maintenance and Checkout

B.24 Transfer: Orbital Transfer is the movement of men (crew), fuel, or equipment from one orbiting vehicle to a second. It will occur between mated or non-mated vehicles, but the vehicles will have to be docked—the relative velocity between the vehicles will have to be zero. This need not occur as an earth orbit maneuver; it could occur in a lunar orbit or in traverse between moon and earth. Transfer of various items will evolve different problems, although many problems will be common to all transfers.

B.25 Assembly: Orbital assembly is the construction of a vehicle, or portion thereof, in space. It can be accomplished by various means: the docking or mating of various assemblies in space resulting in new assemblies, or by the construction in space of assemblies using various manufacturing and assembly techniques. Parts would be delivered to the assembly site by other vehicles. Orbital construction would probably not be considered in the near future missions except for a minor assembly in a vehicle could carry all necessary parts.

B.26 Repair: Orbital repair is the repair in orbit of any equipment or device found to be inoperational or unreliable by orbital checkout, visual observation, or through detection by an alarm system. Orbital maintenance is the servicingin orbit of equipment at prescribed intervals. Items requiring such maintenance would be the environmental control system, such as cooling and heating, nitrogen and oxygen, and pressurization; the vehicle systems, such as power distribution, guidance, control, propulsion, insulation seals, and communications; and life support systems, such as food, liquid, waste disposal, and living quarters. Not only does repair and maintenance apply to own vehicle, but to other vehicles that may dock or rendezvous with the repair vehicle such as in the repair and maintenance of lunar launch vehicles at orbiting launch platforms.

B.27 Checkout: Orbital checkout is the operation by which a component or system is checked for operational readiness. This is usually accomplished by stimulating the system, noting its response, and comparing this with a pre-established value. These values usually have maximum and minimum limits, outside of which the system is considered to be inoperational.

Earth Orbital Launch (Translunar Injection) and Translunar Flight

B.28 Earth orbital launch is concerned with the successful launch of the space vehicle and its boosters from earth orbit through escape velocity toward moon capture or landing trajectory. On initial flights, the space vehicle will launch directly from a parking orbit; on later flights the launch will probably be made from an orbiting launch facility with some type of physical connection between the two. For definitive purposes all orbital sustenance such as fuel transfer, repair, or adjustment will be terminated before final checkout (countdown) and launch. The actual orbital launch will be considered under way as soon as countdown results in booster ignition. The mission will be considered terminated upon injection toward the desired conditions of lunar arrival, whether it be flyby, orbit, or entry for subsequent landing. Translunar flight was not considered as a separate mission because any trade-offs or requirements for midcourse and terminal flight-path correction are directly related to orbital launch accuracy.

B.29 The orbital launch platform is considered economically feasible for repair, adjustment and checkout of the somewhat sensitive and not too stable space craft, instead of throwaway and replacement of entire vehicle stages. The platform is also useful because it will result in a marked decrease in lunar vehicle weight at launch. The power supply and checkout components necessary for lunar launch may remain with the platform. Since earth booster size has restricted the early lunar program to proposed assembly and fuel transfer in orbit, the platform serves as a "workshop" for the completion of these techniques prior to checkout and orbital launch.

B.30 Whether or not there will be physically locked contact between the platform and the vehicle at launch ignition remains to be investigated. Ignition-abort studies on earth have shown that mission reliability rises significantly if physical control (hold down and monitor) can be extended 3 or 4 seconds after main stage ignition. This would be a complex problem while in orbit, but a recovery system to slow and recover the craft should be within the capabilities present should the launch be aborted immediately.

B.31 All problems associated with systems and components will be accuracy problems at orbital launch. An attempt will not be made in this discussion to calculate allowable error for applicable systems at launch. However, it may be noted that the diameter of the moon subtends only 30 minutes of arc when observed from the orbiting platform. Trajectory will involve 240,000 miles line of sight and approximately 60-70 hours flight time. It can be seen that time drift error and prediction error will both contribute significantly to terminal error displacement.

B.32 The orbiting launch platform will be considered to have its ephemeris characteristics well established for initial reorientation of the vehicle inertial system. Any docking impulse to the platform can disturb the prevading momentum and require re-establishment of the platform ephemeris by earth tracking and computation. Several hours are required for this re-evaluation (4 to 5) but the more the better (10 plus), considering the required accuracies involved.

B.33 The final launch window available from earth orbit toward the moon is a very important consideration and is directly concerned with many trade-offs of accuracy versus propulsion capacity requirements. The launch time frame available during 1° of orbit arc traverse in 300 mile orbit is 15 seconds. The final launch window will probably be some fraction of this.

B.34 If the vehicle remains in zero 'g' conditions prior to launch, propellant venting must be monitored to insure that only vapor is vented and that the vents do not provide errant attitude impulse.

Lunar Orbit and Landing

B.35 Lunar Landing is concerned with the actual landing of the space vehicle at a desired point on the surface of the moon. The landing mission will originate from a lunar orbit or possibly from a direct trajectory. Any landing maneuver which directs the space craft toward the surface will initiate the mission. The task is completed when the space craft is at rest at the desired point on the moon's surface.

B.36 There are several problem areas associated with such a mission. First, assuming that there has been adequate intelligence acquired about the moon crust, to pick a landing field and several alternate sites, or at least to give the crew a measurable criterion for picking their own site, there will quite surely be large circular error about any target area which requires correction to facilitate landing. This demands the space craft to be capable of displacing more-than-negligible lateral distance while in a reverse configuration and quite possible applying full retro thrust. All these capabilities operating concurrently while in a "back-down" configuration place high demands on the guidance, control, and propulsion system.

B.37 Whether or not a human link is to be included in the terminal landing sequence is unknown at this time; a TV landing display would serve as visual back-up for terminal landing if the structure of the crust is not of such a dusty character that it would attenuate all lunar or electronic observation upon terminal landing sequence.

B.38 The problems involved in landing a manned vehicle vertically on unknown and unimproved terrain could be critical and will require capable fact-finding in most technical areas covered in the study.

Lunar Launch and Transearth Flight

B.39 The goal of the lunar launch mission will be to successfully inject the space craft into earth-bound trajectory for platform rendezvous or re-entry orbit and surface landing. The requirements for and the type of lunar launch depend on the intended mission of the vehicle. Figure B.1 shows the various lunar launch capacities required in terms of a particular lunar goal. All three lunar goals will be attempted at different times in the manned-lunar program. For the present, at least, the possibility of lunar orbital rendezvous will be considered as beyond the expectations of the present program. Examination of Figure B.1 shows that lunar flyby or circumlunar flight are the least demanding of the three goals in terms of an actual lunar launch. Manned flyby will be discussed only to the extent that there will be probable trajectory and velocity corrections taking place which are considered to be the extent of "launch" requirements.

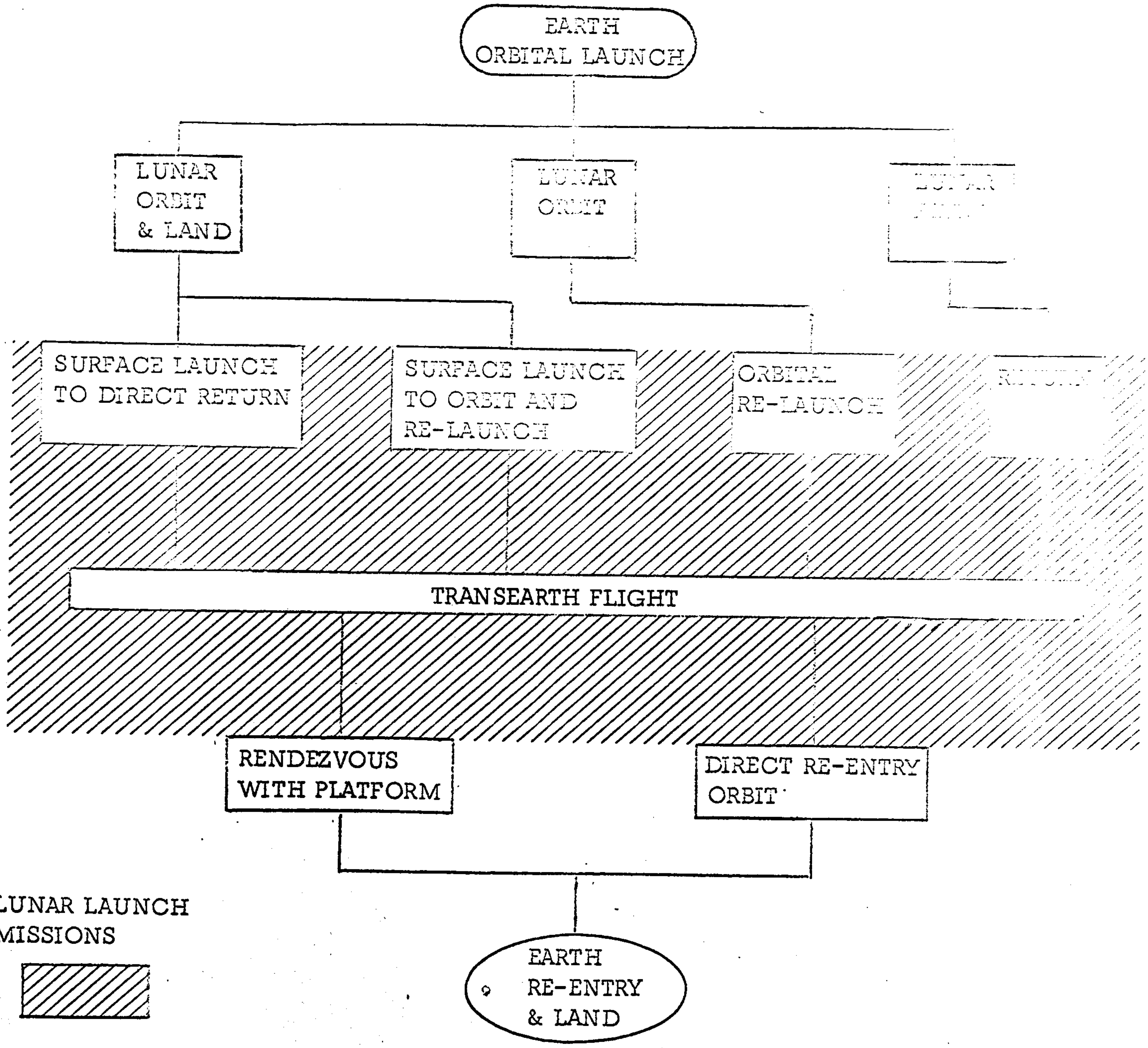

FIGURE B.1. LUNAR MISSION SEQUENCES

B.40 Lunar orbit re-injection toward earth will be a major characteristic of the two remaining programs. This operation will be quite similar to the earth orbital launch only without the support of the orbiting launch platform and the ground support tracking complex. All components to determine initial orbital launch predictions will have to be carried on board and subject to the 60-70 hour flight environment from earth. In this mission, accuracy requirements become almost overwhelming and the necessity for reliable accuracy trade-off studies are great.

B.41 The remaining lunar launch possibility is launch from the surface of the moon to direct trajectory or lunar orbit. For the time being, this discussion will not include requirements arising from a choice of either flight path, but only of general area problems involved in launch itself.

B.42 Lack of information pertaining to the surface stability and composition of the moon crust allow imaginative reflection upon requirements of a 50-75,000 lb (earth weight) space craft at rest on the surface. Whether the space crew could be called on to erect any primitive launch pad remains to be investigated.

B.43 Guidance systems should be realigned prior to launch. Since the probable circular landing error on the moon will be quite large, some type of geographical or spacial orientation will have to be established for moon to earth guidance prediction computation. The remaining general problems will be similar to those associated with earth launch although complicated by the primitive environment and stringent accuracy requirements. The problems arising from mechanical, acoustical, and thermal stress at take-off will undoubtedly be less severe than at earth launch and should not present particular difficulties at that time.

B.44 The lunar launch mission will terminate when thrust is expended by the space craft as it nears the earth in order to impinge toward earth orbit for rendezvous or re-entry maneuvers.

B.45 As the space craft returns to earth it has the choice of direct and immediate re-entry or assuming an earth orbital course. Once the orbital course is chosen, the craft may orbit alone prior to ballistic re-entry or may attempt rendezvous with the still orbiting launch platform.

B.46 The simplest, but most adverse, course is direct re-entry. Manned survival would seem to be questionable after considering direct impact with the earth's atmosphere at approximately moon escape velocity, also target circular errors would be so large that capsule recovery might require a monumental coverage of part of the earth's surface.

B.47 Assumption or orbit would require some thrust capability to retro and drop into chosen orbit attitude. Then a final retro thrust capability to drop into re-entry at the proper time similar to the proposed Mercury capsule flight program. It is not proposed to calculate energy requirements for re-entry paths at this time, but an approximate negative velocity change would be required to retro into earth orbit upon termination of moon-earth inter-orbital flight. Timing and guidance requirements would not be necessarily so strict as for either direct re-entry or orbital rendezvous.

B.48 As mentioned previously, inbound rendezvous requirements are most severe since the maneuver involves matching earth orbital planes and orbital velocities.

B.49 The terminal flight path of the space craft must be rigorously analyzed for accuracy/reliability/propulsion capacity trade-offs. It is a very critical area in terms of success of the entire man-lunar mission.

Earth Re-entry and Land

B.50 Re-entry of a lunar mission launch vehicle is the return of the vehicle into the earth's atmosphere from earth orbit, an earth orbiting launch platform, or from the moon following transearth flight. It includes the orientation and control of the launch vehicle in the earth atmosphere such that the re-entry and subsequent landing will occur at the desired location at the desired time.

B.51 Re-entry results from propelling the launch vehicle toward the earth, or by slowing down the vehicle to below orbital velocity as it orbits or passes by earth, such that the vehicle will be attracted to the earth by gravity. Re-entry begins when the vehicle is earth bound and obligated to land; it ends with successful landing of the vehicle.

B.52 A re-entry vehicle may be nothing more than a manned ballistic space capsule relying entirely on gravity for earth bound propulsion or it may be a self-propelled space vehicle capable of extensive flight within the earth's atmosphere until landing is desired. In first generation vehicles ballistic techniques will be used; an L/D of 0.5 has been designated for the Apollo vehicle.

B.53 Before re-entry is initiated, a complete checkout of the vehicle systems to be utilized in re-entry should be conducted prior to release from the earth orbiting launch platform or during the translunar flight, depending on the route of return travel. A continued monitoring of the earth's movements will have to be maintained in order to ascertain the most desirable re-entry characteristics and tactics. If re-entry is to occur following flight from the moon, maneuvering of the vehicle to attain the desired re-entry corridor will take place during this long flight (240,000 nautical miles/60-70 hours). An important consideration in this maneuvering will be the conservation of fuel. Monitoring of the earth's movements would be a responsibility of the orbiting launch platform if re-entry is to occur from this vehicle.